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PRELIMINARY WEIGHT AND COST ESTIMATES FOR TRANSPORT AIRCRAFT COMPOSITE STRUCTURAL DESIGN CONCEPTS

Prepared under Contract No. NAS1-10702 by

GENERAL DYNAMICS
Convair Aerospace Division
Fort Worth Operation

For NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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ABSTRACT

Preliminary weight and cost estimates have been prepared for design concepts utilized for a transonic long range transport airframe with extensive applications of advanced composite materials. The design concepts, manufacturing approach, and anticipated details of manufacturing cost reflected in the composite airframe are substantially different from those found in conventional metal structure and offer further evidence of the advantages of advanced composite materials.

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LIST OF SYMBOLS

AR	aspect ratio
B_N	maximum nacelle breadth
b'	structural span
$\overline{c}$	wing mean geometric chord
D	fuselage diameter
D_n	maximum nacelle depth
F_{tu}	ultimate tensile stress
F_H	horizontal tail balancing load
F_V	vertical tail load
H	fuselage height
L_t	length of structure over which a load is assumed to act
M	Mach number
MAC	Mean Aerodynamic Chord
S_{HT}	horizontal tail area
S_N	nacelle cowl surface area
S_{PYL}	pylon planform area
S_{VT}	vertical tail area
S_W	theoretical wing area
S_{WETT}	fuselage wetted area
T	maximum operating temperature
t/c	wing thickness to chord ratio

LIST OF SYMBOLS (Cont'd)

W_{DES}	flight design gross weight
W_{NC}	weight of nacelle contents (excluding nacelle structure)
λ	planform taper ratio (tip chord/root chord)
Λ	sweep angle

SECTION 1

SUMMARY

Additional studies beyond the Advanced Transport Technology system study programs have been conducted in an effort to provide additional information about the design concepts, structural weights, manufacturing approach, and potential manufacturing costs anticipated for the transonic long-range transport which has extensive applications of advanced composite materials. Design concepts are shown for most of the major structural areas and detail weight statements have been prepared. Weight statement summaries show that 68 percent of the airframe is composed of composite materials, 11 percent is honeycomb core and adhesive, and 21 percent is aluminum, titanium, or steel. Alternate concepts were also evaluated.

The manufacturing approach makes extensive use of automated and mechanized equipment for the fabrication of the very large sections required. The approach to manufacturing the composite transport will be substantially different from that utilized in the manufacture of conventional metal aircraft.

Detail cost analyses have been prepared for the manufacturing activities required. These cost analyses have supported earlier findings which identified significant cost benefits through the use of composite structure in transport aircraft.

SECTION 2

INTRODUCTION

Design and economic studies conducted as authorized by NASA Contract NAS1-10702 and reported in Reference 2-1 have described the potential weight and cost savings associated with the application of advanced technologies to transport aircraft. This report describes the findings of subsequent studies conducted to provide additional preliminary information about the design concepts, structural weights, manufacturing approaches, and anticipated costs of transport aircraft structure fabricated with advanced composite materials.

The transport aircraft configuration used as a baseline in these studies was the Mach 0.98 configuration originally presented in Reference 2-1 and described again in this section for ease of reference. A general arrangement of the Mach 0.98 configuration is presented in Figure 2-1. Overall dimensions of the aircraft are: length, 193 ft. 7 in. (59 m); height, 51 ft. 1.5 in. (15.58 m); span, 141 ft. 9 in. (43.21 m). The fuselage is area-ruled, and the wing, empennage, and engines are arranged to achieve the optimum cross-sectional area distribution for an aircraft cruising at Mach 0.98 airspeed.

Maximum fuselage diameter is 18 ft. 11 in. (5.77 m). This allows seven-abreast seating in the coach section and the carriage of standard LD-3 containers in the cargo bay. The necked-down mid-section has a minimum diameter of 13 ft. 7 in. (4.14 m) and provides five-abreast seating.

The supercritical wing has an 8.0 aspect ratio and is swept at the mid-chord to 40 deg. The t/c at right angles to the mid-chord measured at the mean chord location is 11.0 percent. Two engines are mounted beneath the wing on pylons and the third is mounted above the fuselage in the base of the vertical tail. High lift is obtained by inboard and mid-span double-slotted Fowler flaps, simple outboard flaps, and Kruger-type "Varicam" leading-edge slats.

The studies reported here were concerned only with the design and manufacture of the transport airframe. Subsystems costs were not considered nor were aircraft operating costs.

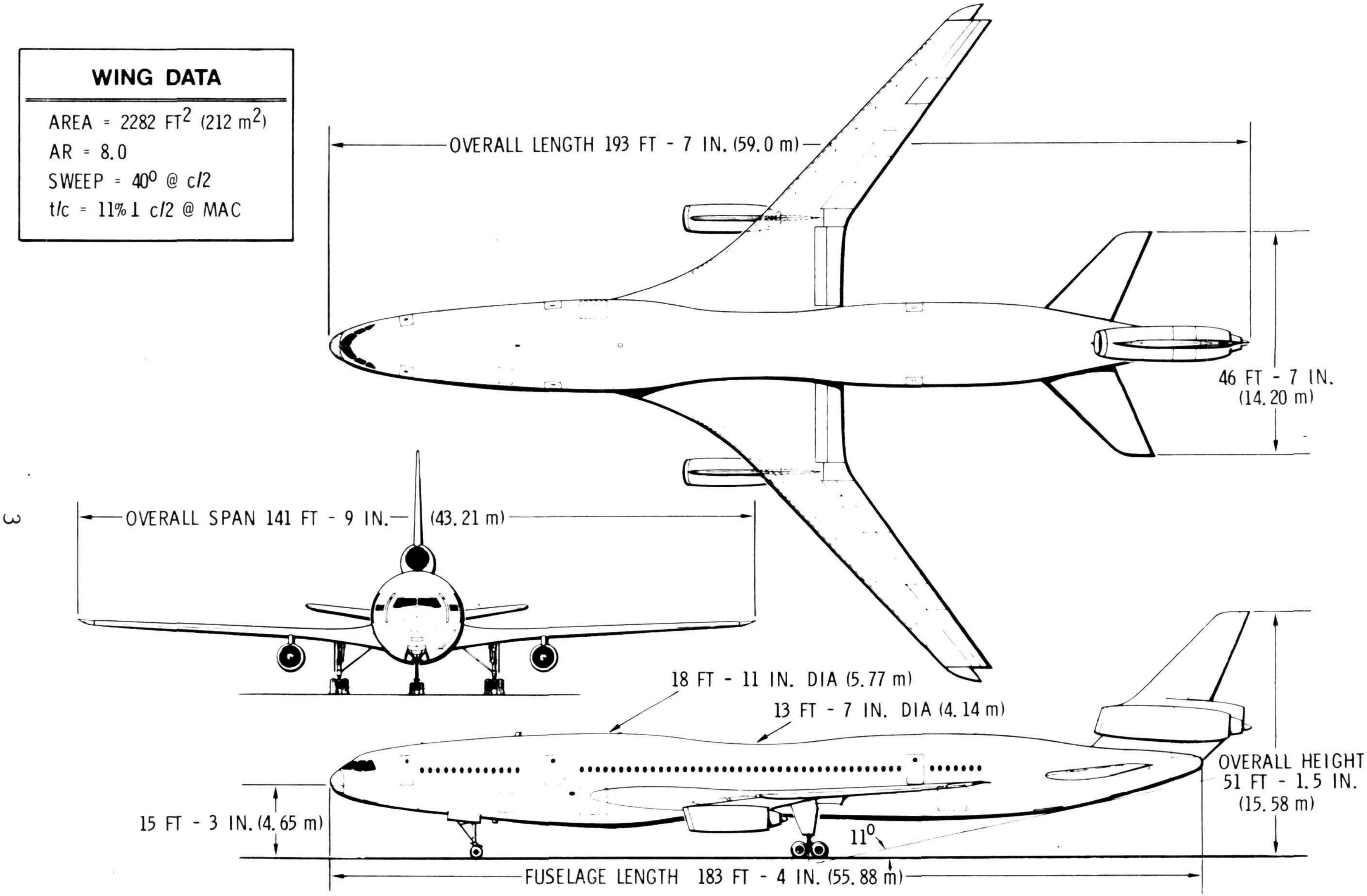


Figure 2-1 General Arrangement, Mach 0.98 Configuration

The following sections present discussions of the design features and weight estimates of the airframe components, the manufacturing approaches for these components, and the anticipated manufacturing costs of an assumed 250 aircraft production program.

Measurement values contained in this report are in both customary and systeme internationale (SI) units with the former stated first and the latter in parentheses. The principal measurements and calculations have been made in the customary system of units.

SECTION 3

DESIGN STUDIES

Design studies were conducted to provide additional information concerning the airframe structural concepts of an advanced composite long-range transport. The studies produced engineering drawings and layouts of many of the design features of the composite airframe configuration. In addition, detail weight breakdowns of the structure were computed. Both design concepts and weight summaries are discussed in the following paragraphs.

3.1 STRUCTURAL DESIGN

After selecting the materials, design details were developed for the wing, fuselage, horizontal and vertical tails, and the nacelles. Alternate structural concepts were also evaluated and assessments were made of thermal and acoustic considerations on the selection of fuselage design concepts.

3.1.1 Material Selection

The composite material selected for evaluation in this design and cost study was a graphite-fiber-reinforced plastic. The plastic could be either epoxy or polyester. Graphite reinforcement was selected over boron because of an anticipated lower total aircraft system cost. Although boron preceded graphite in the technology development, many of the structural graphite-reinforced materials are already lower in cost than the boron, with even lower graphite costs predicted as sales volume continues to increase. A second source of lower total costs with graphite is in the manufacture of hardware. Graphite preimpregnated materials are more easily shaped to complex contours during lay-up and are less difficult to machine in cured shapes so that the manufacturing costs are lower than those resulting from use of boron-reinforced materials. Glass-fiber-reinforced plastic is used because of the low cost in local areas where its mechanical properties are acceptable and good formability is required.

The graphite or glass-fiber-reinforced preimpregnated materials assumed available in this study will develop acceptable mechanical properties when cured at a temperature

not exceeding 250°F (394°K) in a vacuum-bag environment and over a time span not exceeding three hours. The tape material for large parts such as wing and fuselage skins will be 24 in. (.61 m) wide, weighing approximately 165 lb (74.6 kg) in a roll 1320 ft (402 m) in length. The material will be delivered to the work area daily without having to undergo a low-temperature, extended storage stage.

3.1.2 Wing

Wing structure consists of a continuous structural box from tip to tip, leading-edge and trailing-edge movable control surfaces, and fixed wing secondary structure. The wing structural arrangement is shown in Figure 3.1-1.

3.1.2.1 Primary Structure

The primary structural box consists of upper and lower sandwich skins, two sandwich spars, and ribs placed at appropriate spanwise locations. Wing skins are of honeycomb sandwich construction with graphite composite facings over aluminum honeycomb core. Skins are manufactured in one piece, tip to tip and front spar to rear spar. The outer facing is of constant basic thickness overall. A minimum number of buildups are used, and these are co-located on the core side with a core splice as shown in Figure 3.1-2. This eliminates the machining of steps and allows sections of core to be of constant thickness. All other skin thickness variations are incorporated in the inner facing away from the core. The result is a one-piece continuous skin with no splices. Both skins are mechanically attached to the spars as shown in Figure 3.1-2. Manufacturing and cost considerations lead to the conclusion that a sandwich box cover is better than the sheet-stringer approach although the shifting of the effective bending material away from the extreme edge results in additional weight. In addition, fabrication cost considerations also lead to a desire for a constant thickness core and constant thickness external facing in the sandwich. This feature is desirable even though the use of a variable thickness outer skin is more weight efficient. The assembly difficulties of the ribs and fuel bulkheads mating with the spanwise stiffeners also favor the sandwich skin approach.

Continuous wing skin, in addition to manufacturing and cost advantages, also offers structural endurance and possible failure safety in the wing fuselage intersection design. The continuous skin provides the load path for the bending moment

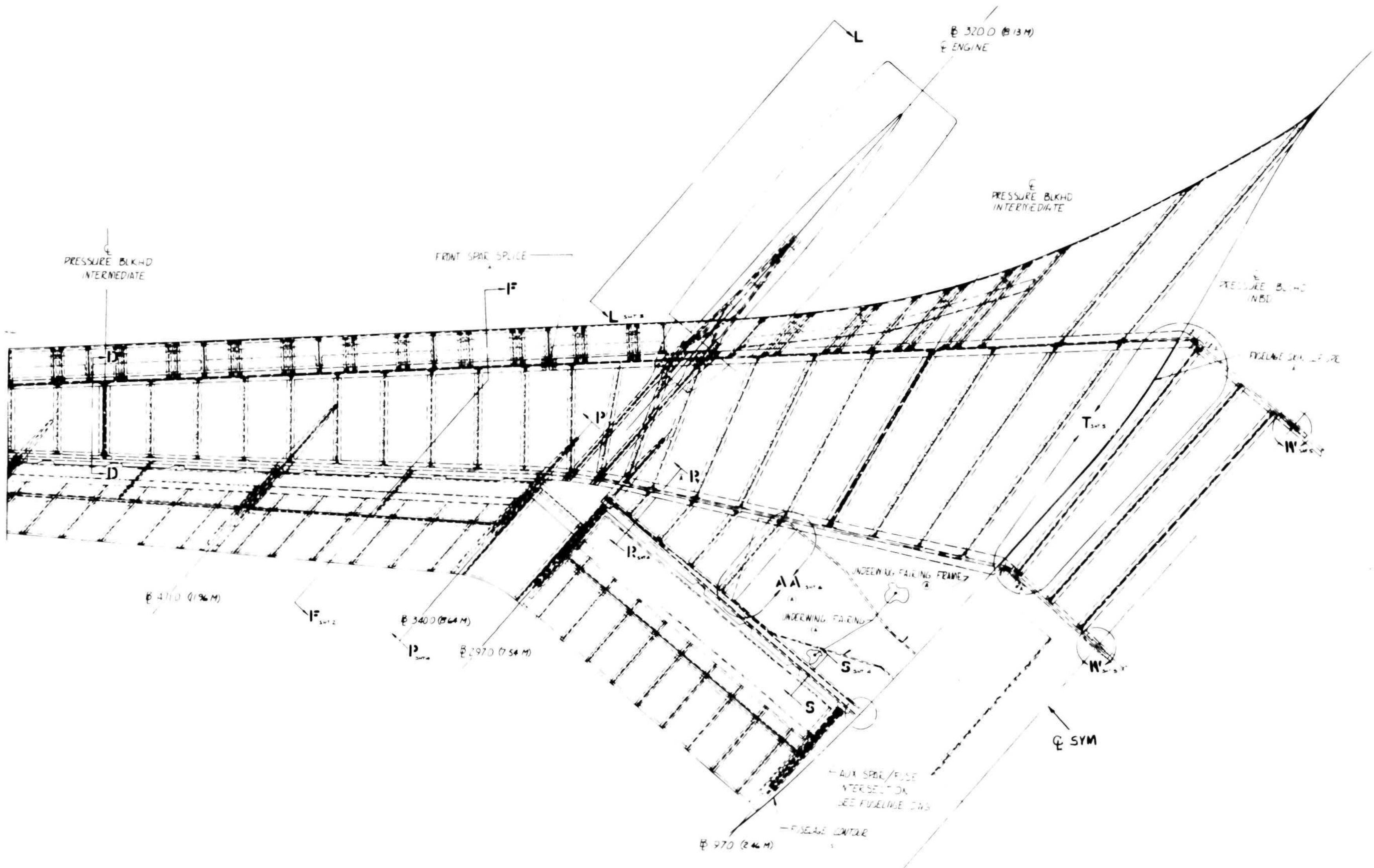


Figure 3.1-1 Wing Structural Arrangement

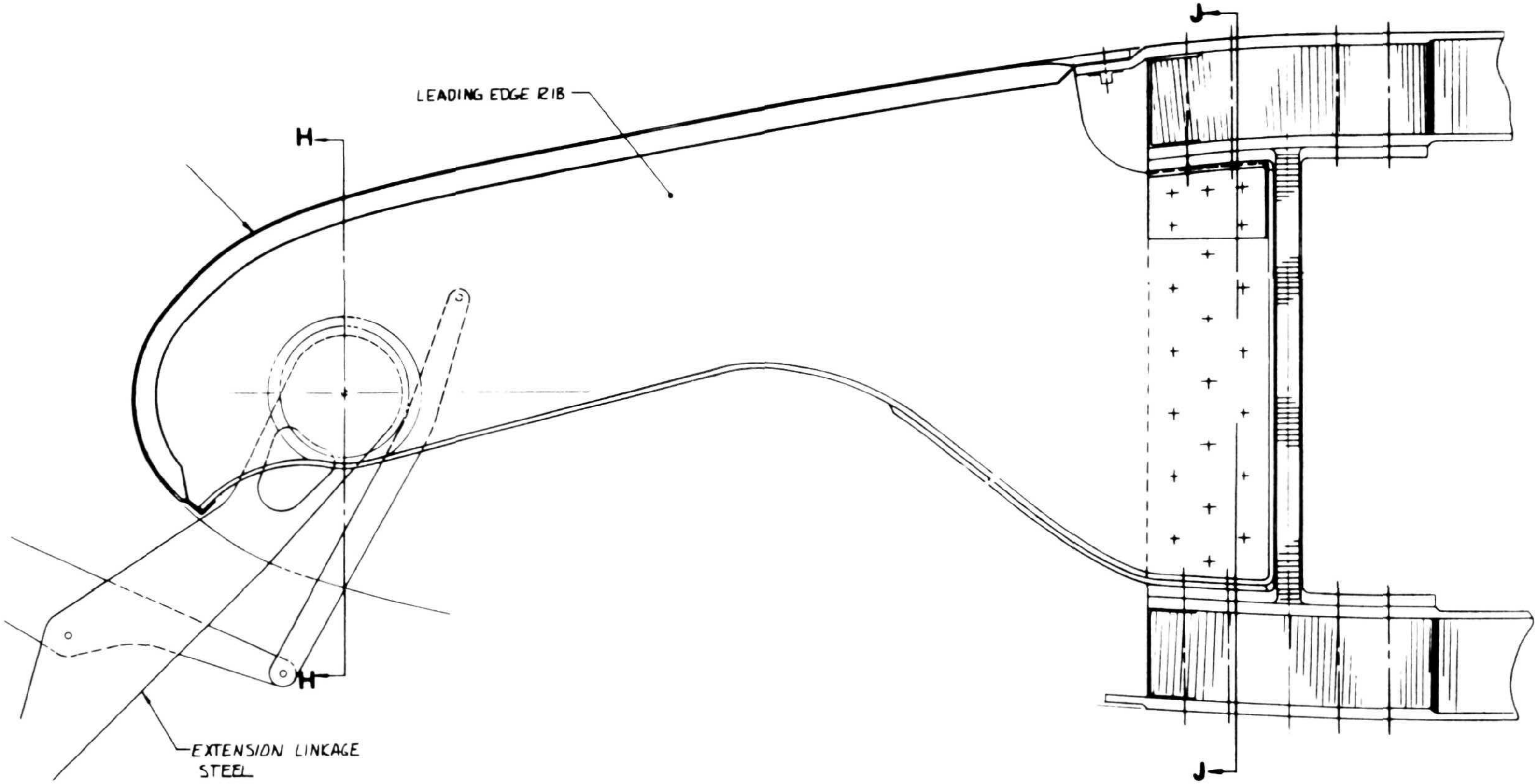


Figure 3.1-2 Skin Buildups

so that only the vertical and horizontal shear and torsional moment require a fuselage reaction. The vertical shear is introduced into the fuselage with a spar-to-bulkhead mechanical attachment, and the horizontal shear is transferred from the wing skin to fuselage skin with a similar mechanical fastener arrangement. The wing-fuselage intersection is shown in Figure 3.1-3.

Two spars form the front and rear boundaries of the wing box. The front spar is located at the 15 percent theoretical root chord and 28 percent tip chord while the rear spar is at the 60 percent chord line outboard of the expanded root and forward of the main landing gear bay inboard. The front spar is made in four sections with a splice on the centerline of the airplane and a splice outboard of the engine pylon. The rear spar is manufactured in two pieces with the splice on the aircraft centerline and has a large radius bend beginning near the engine pylon centerline. This spar splice arrangement is changed from the previous study reported in Reference 2-1 to eliminate the spar splice in the already complex wing-fuselage intersection area. The outboard splice was added to the front spar to facilitate assembly. Spars are made of two graphite-composite channels bonded back-to-back with aluminum honeycomb core between them as shown in Figure 3.1-2. The core is constant thickness and all web laminate thickness variations are away from the core. Access doors for wing assembly, maintenance, and inspection are located in the front spar as shown in Figure 3.1-4. The auxiliary spar is straight and intersects the aft spar in the engine pylon attach area as shown in Figure 3.1-1. It is fabricated of two graphite composite channels located back to back and attached to graphite composite laminate caps to form a box section as shown in Figure 3.1-5. The auxiliary spar supports the aft end of main landing gear beam which is a graphite composite rectangular section as shown in Figure 3.1-6. Design of a beam with sufficient depth to carry the aft auxiliary spar loads required the lower wing contour to extend below the theoretical contour as shown in Figure 3.1-6. An underwing fairing was added to form the wing lower aerodynamic surface in the auxiliary spar and main landing gear beam area.

Ribs are located at approximately 30-inch (0.76 m) intervals and normal to the 60 percent chord outboard of the engine pylon. Inboard of the engine pylon the ribs become nearly streamwise as shown in Figure 3.1-1. Ribs are graphite-composite sheet with fiberglass composite stiffeners. The

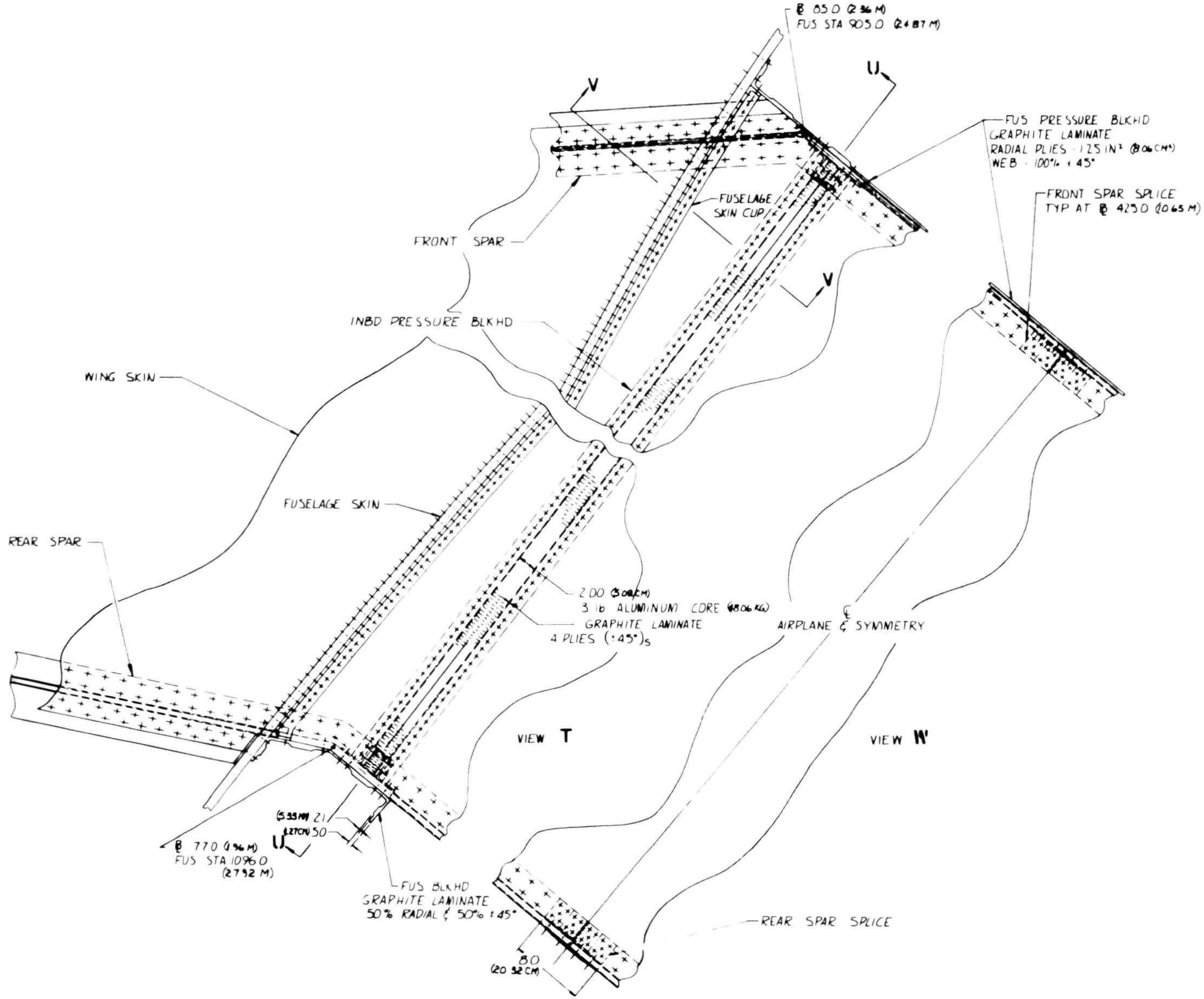


Figure 3.1-3 Wing-Fuselage Intersection

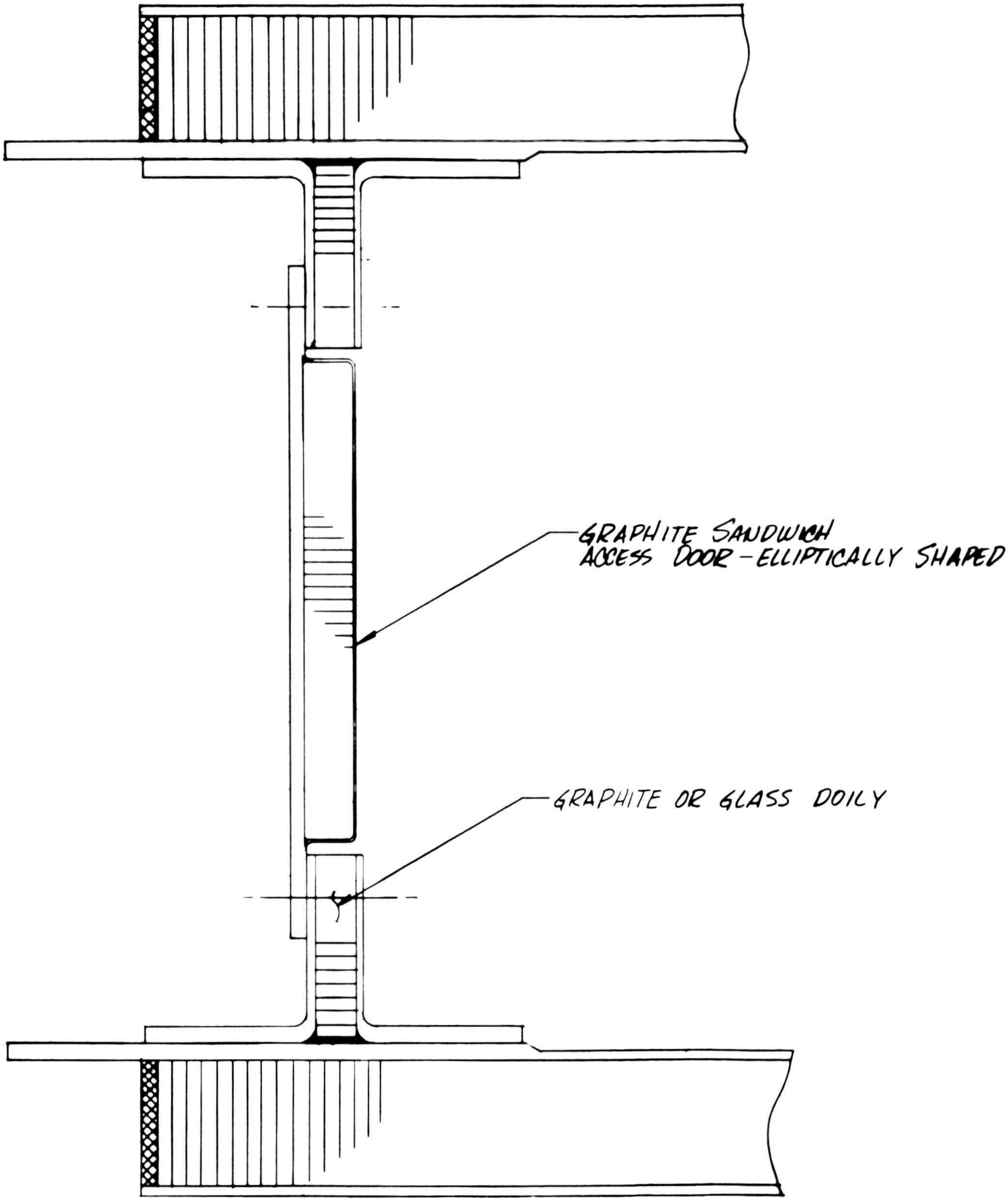


Figure 3.1-4 Access Door in Front Spar (TYP.)

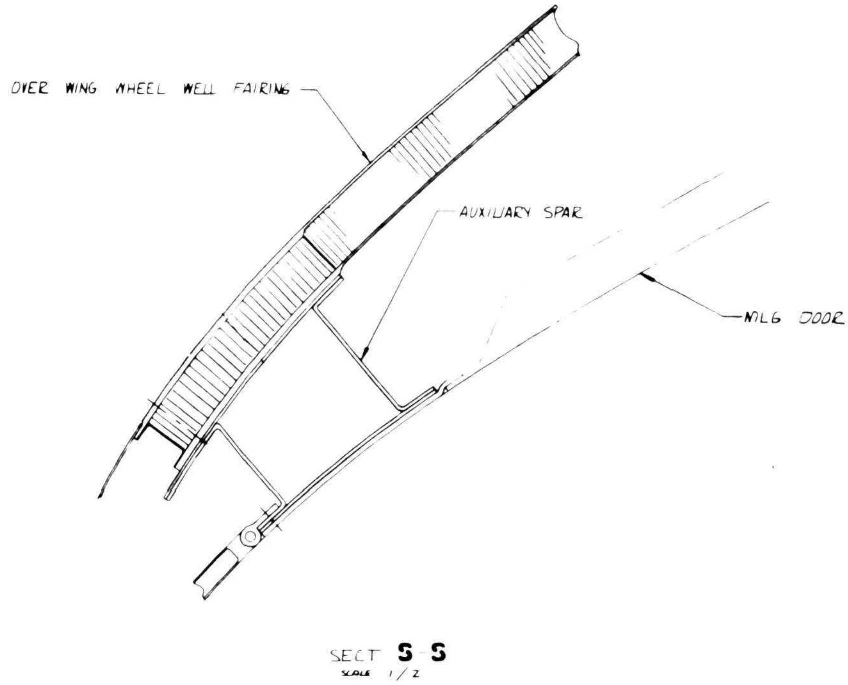


Figure 3.1-5 Auxiliary Spar

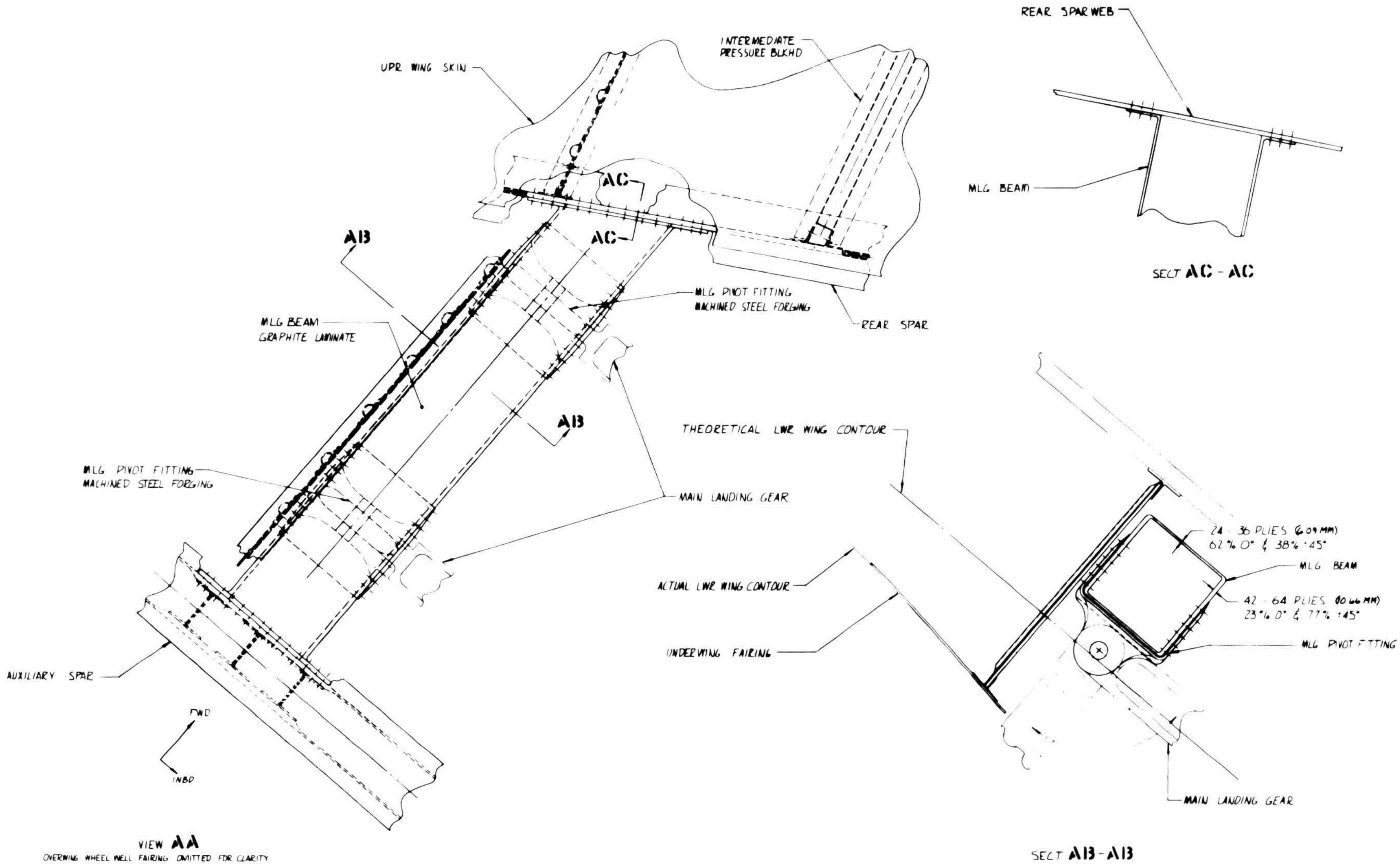


Figure 3.1-6 Landing Gear Beam

ribs are bonded to the upper and lower skins and rear spar and mechanically attached to the front spar.

The fuel pressure bulkheads are sandwich panels with graphite-composite face skins and aluminum honeycomb core, and are mechanically attached to the spars and skins. The rib and fuel pressure bulkhead attachments are shown in Figure 3.1-7.

3.1.2.2 Wing Control Surface Structure

Control surfaces consist of the trailing edge flaps, vanes, ailerons, spoilers, leading edge variable cambered slats, and wing tip flutter suppressors.

The flaps and aileron segments are sandwich construction with graphite composite facings and aluminum honeycomb core. The single front spar and chordwise ribs are graphite-composite sheet with fiberglass shear ties to the sandwich skins as shown in Figure 3.1-8. A graphite-composite closure channel forms the blunt trailing edge typical of the super-critical airfoil.

Vane skins are of sandwich construction with graphite-composite outer and inner face skins, and multiflex honeycomb core. A graphite-composite hat section channel forms the understructure of the vane. The feet of the channel are bonded to the lower skin, and the top is bonded to the upper skin. The trailing edge is constructed as a graphite laminate and is mechanically attached to the other portion of the vane. All end closure members are graphite-composite sheet stiffened as required with fiberglass.

Spoiler segments are of full depth bonded honeycomb sandwich. Sandwich facings are graphite-composite laminates, and the core and support fittings are aluminum. The same type of construction is used for the air deflection doors.

Flap tracks and highly stressed fittings are steel or titanium. Aileron support fittings, spoiler hinges, and actuator attach fittings and others which are less highly stressed are aluminum.

Leading edge "varicam" slats consist of a membrane sheet member forming the undersurface of the wing in the stowed position, a stiff nose piece, and a linkage that actuates the slat and also deforms it to the required shape

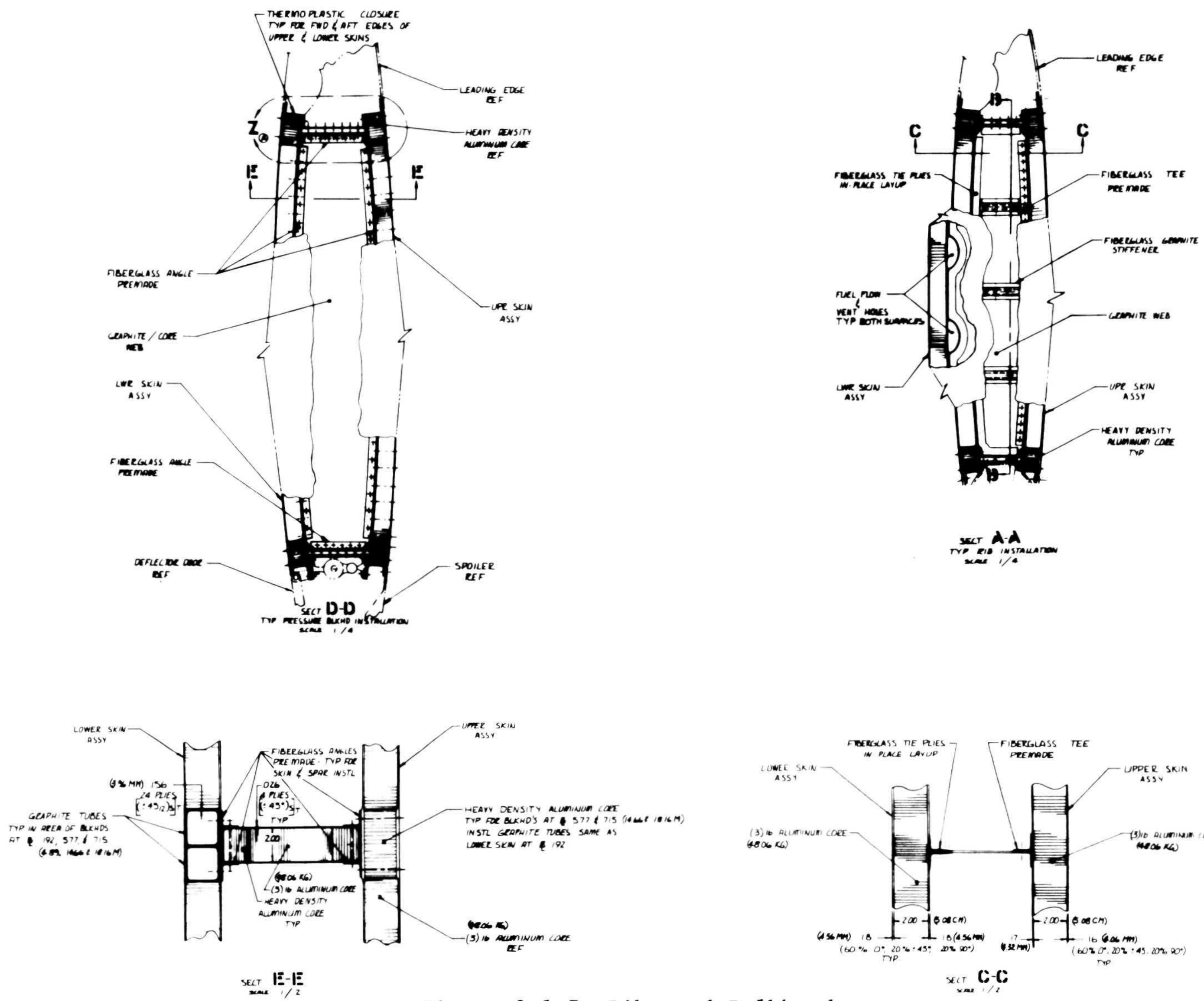


Figure 3.1-7 Ribs and Bulkheads

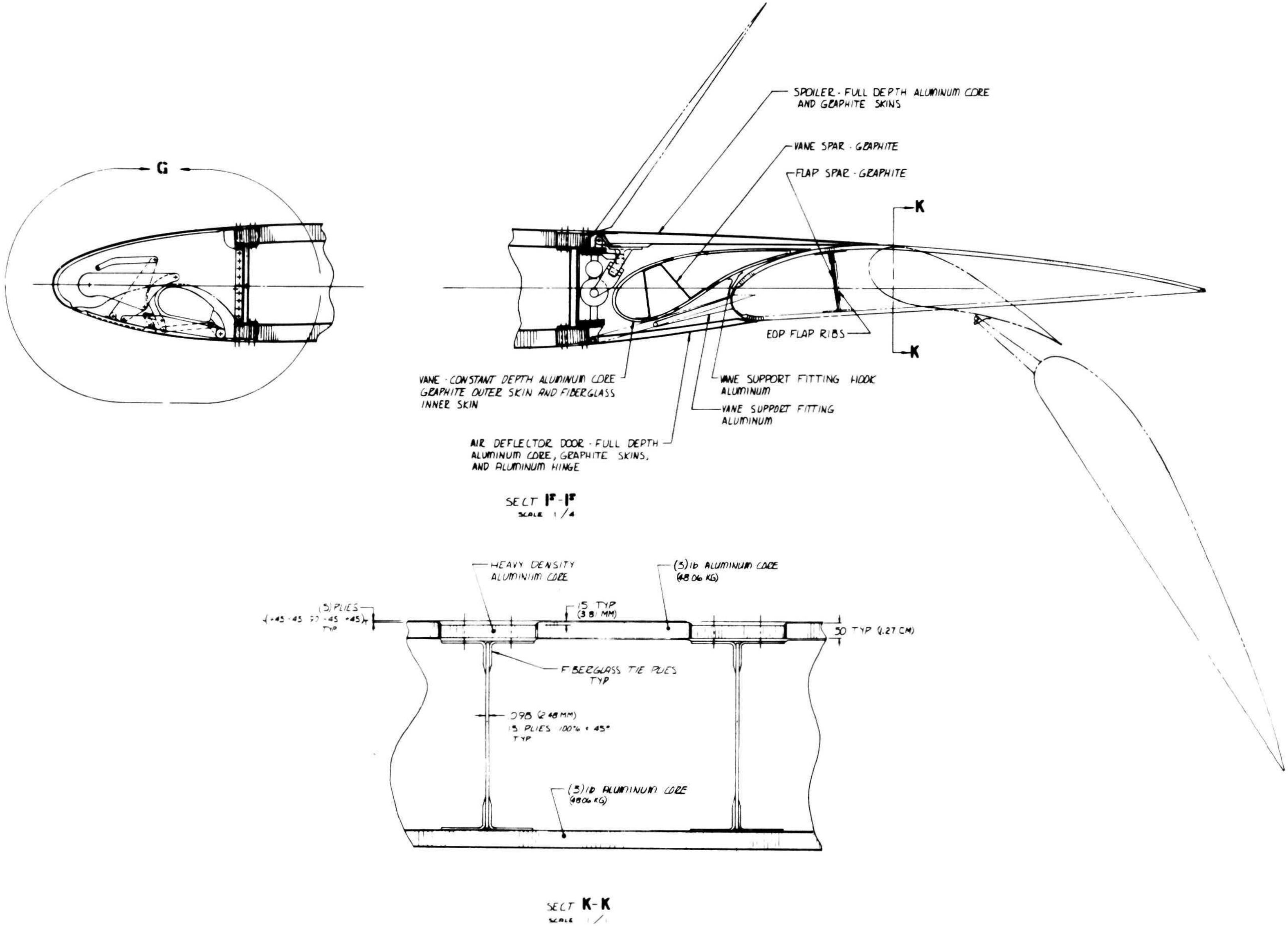


Figure 3.1-8 Control Surface Structure

as shown in Figure 3.1-9. The membrane sheet is fabricated of fiberglass reinforced in the spanwise direction with graphite-composite fibers. The slat nose is made of a thermoplastic extrusion bonded to the understructure as shown. A metal piano hinge fitting attaching the nose segment to the sheet skin is bonded into the nose segment. The fixed leading edge is a graphite sheet with integral core stiffeners. The mechanical linkage consists of a graphite torque tube with metal attaching links.

The wing tip flutter suppressor is attached to the fixed tip steel rib with a pivot mechanism as shown in Figure 3.1-10. The tip structure is graphite laminate skins over aluminum honeycomb core.

3.1.2.3 Fixed Secondary Structure

The fixed leading edges and fairings are generally graphite-composite laminates or sandwich structure using graphite-composite facings and aluminum honeycomb core. Understructure is graphite-composite laminated sheet with fiberglass stiffening. The components are permanently assembled by bonding and riveting but are installed on the wing using mechanical fasteners where they may be removed and replaced if damaged.

3.1.3 Fuselage

The fuselage structure consists of three principal components. These are (1) the nose structure containing the flight deck and electronics bay, (2) the main cabin area consisting of the passenger compartment, cargo compartment and wheel wells, and (3) the tail structure consisting of the empennage support structure and auxiliary power unit bay.

3.1.3.1 External Shell Structure

The external shell is bonded sandwich structure and is fabricated from 1 in. (0.025 m) minimum honeycomb core and graphite-composite laminated facings. The composite facing laiminates are thickness tailored and the fibers are directionally oriented for optimum structural efficiency. The shell structure is designed to be manufactured in three sections, each the full length of the cabin and about 120 degrees (2.1 Rad.) of the circumference. Frames are hat shaped sections of fiberglass fabric with graphite reinforcing plies added to the cap as shown in Figure 3.1-11. The ring frame

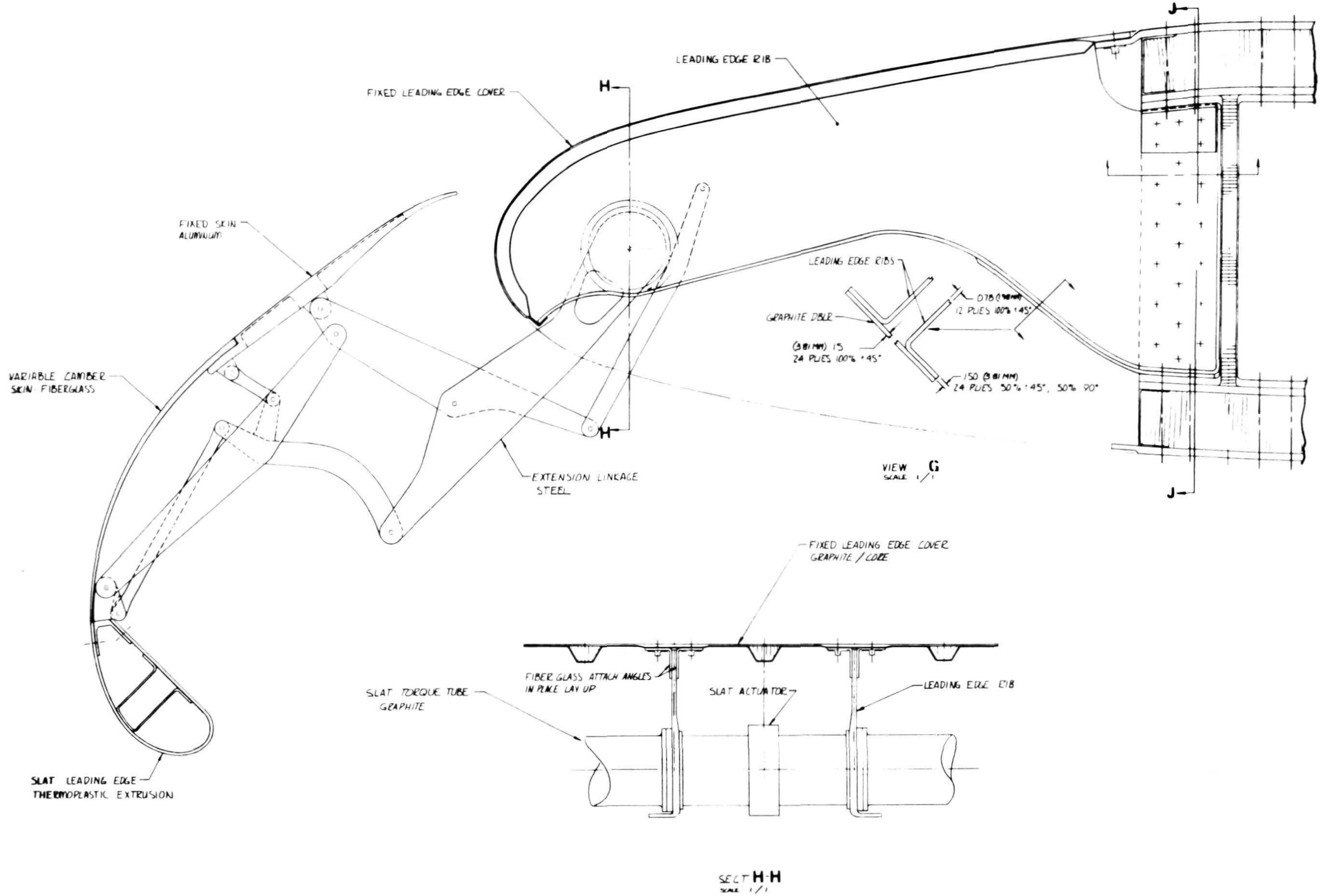


Figure 3.1-9 Leading Edge Slat

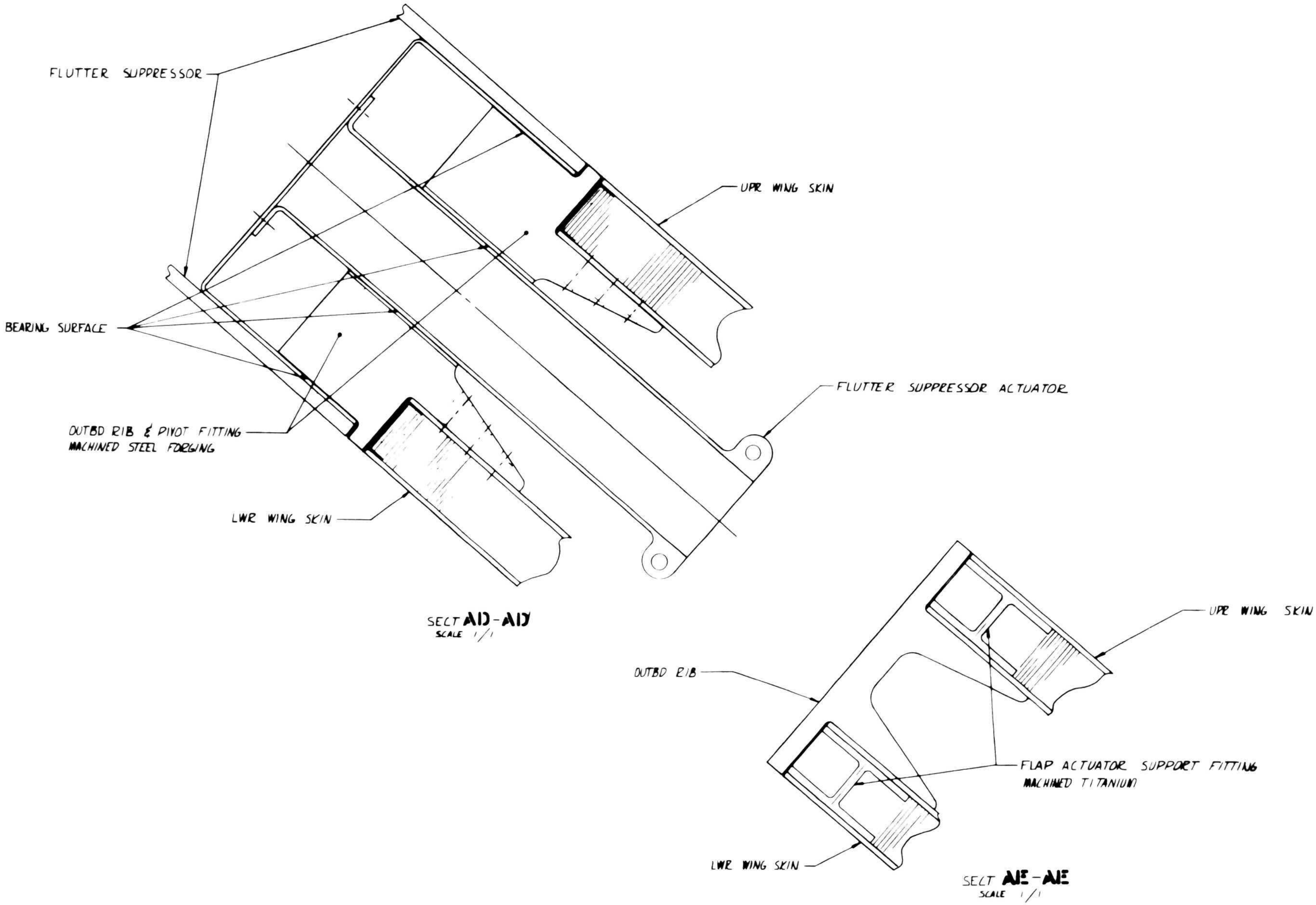


Figure 3.1-10 Tip Flutter Suppressor

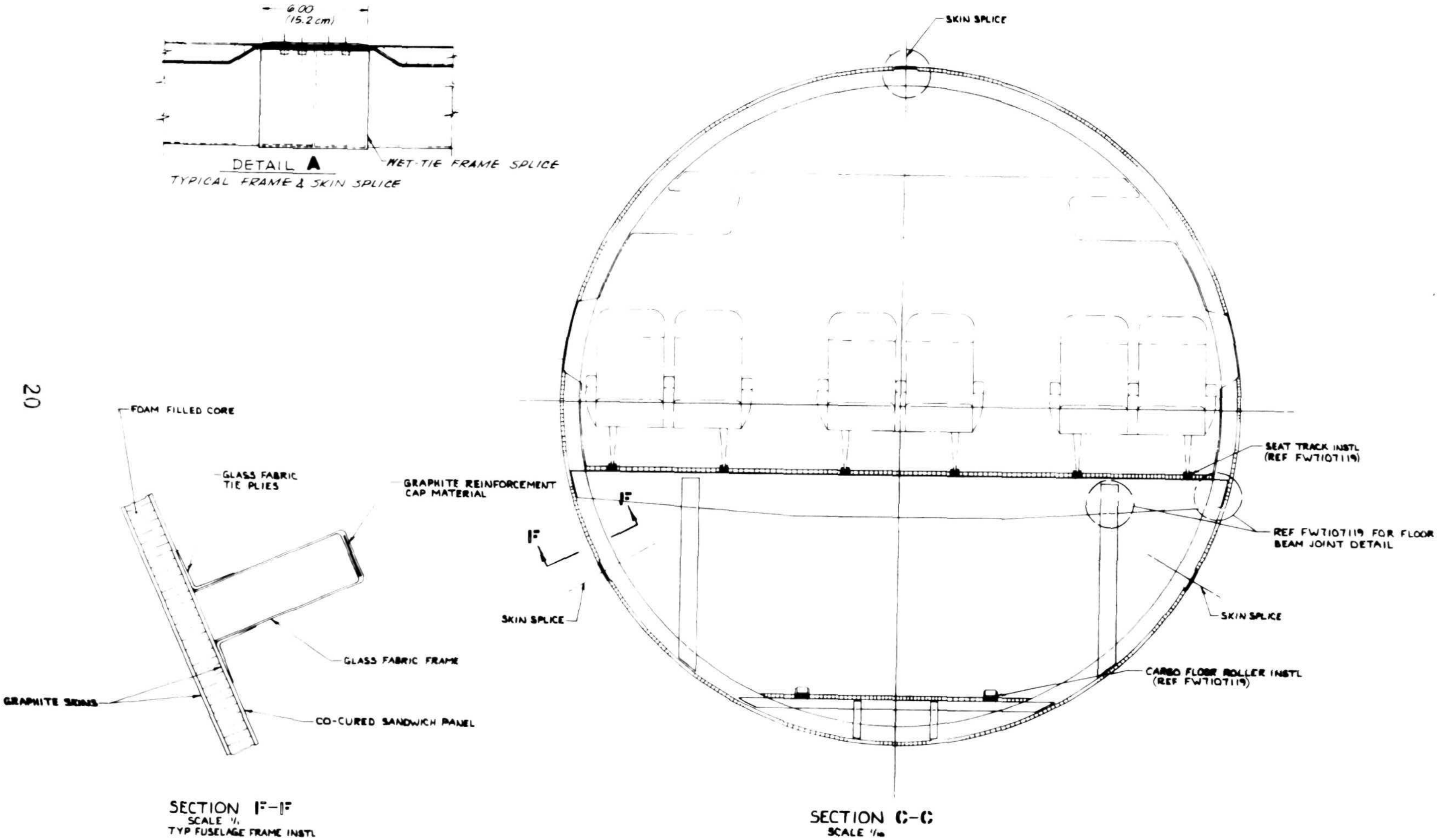


Figure 3.1-11 Fuselage Frame

segments are bonded to the skin panels prior to final assembly. The bond attachment at approximately every fourth frame and in other local areas is reinforced with fasteners through the frame flange and inner skin face for added strength. The shell subassemblies are joined at final assembly by splicing at the upper centerline and at 60 degrees (1.05 Rad.) from the lower centerline. External and internal skin splices are bonded in place and reinforced with mechanical fasteners in double shear. Frame segments are spliced using a bonded wet layup graphite-composite splice as shown in Figure 3.1-12.

The core in the upper two fuselage panels is omitted in the window belt area. Appreciably thicker basic skin laminate is required due to the large number of window cutouts. Each cutout is elliptically shaped to minimize hoop tension stress concentrations as compared to a square or rectangular opening, and offers more window area than a comparable round opening. A composite doubler and a pan assembly are added around each cutout and are bonded in place as shown in Figure 3.1-13.

3.1.3.2 Internal Fuselage Structure

Cabin floor transverse beams are attached to each fuselage ring frame. These are formed graphite-composite laminate channels stiffened with fiberglass hat stiffeners bonded to the beam web. Cargo floor beams are fabricated essentially the same.

In Reference 2-1, longitudinal floor beams were substituted for the transverse beams in the overwing area. Additional study revealed that wing deflections were not of a magnitude to cause structural degradation or passenger awareness and that by continuing the transverse beams in this area, large load concentrations into the fuselage skin would be eliminated.

Cabin floor panels are of sandwich construction using graphite-composite facings and either aluminum honeycomb or edge-grain balsa wood core depending upon the floor usage. Floor panels are installed directly on the floor beams with the seat tracks attached through laminate strips in the floor panels as shown in Figures 3.1-11 and 3.1-12. Cargo floor panels are installed in a similar manner. This method of floor panel installation resulted in a considerable weight and cost savings over the method reported in Reference 2-1 of installing floor panels between seat tracks and cargo roller rails and not supporting them directly by the floor beams.

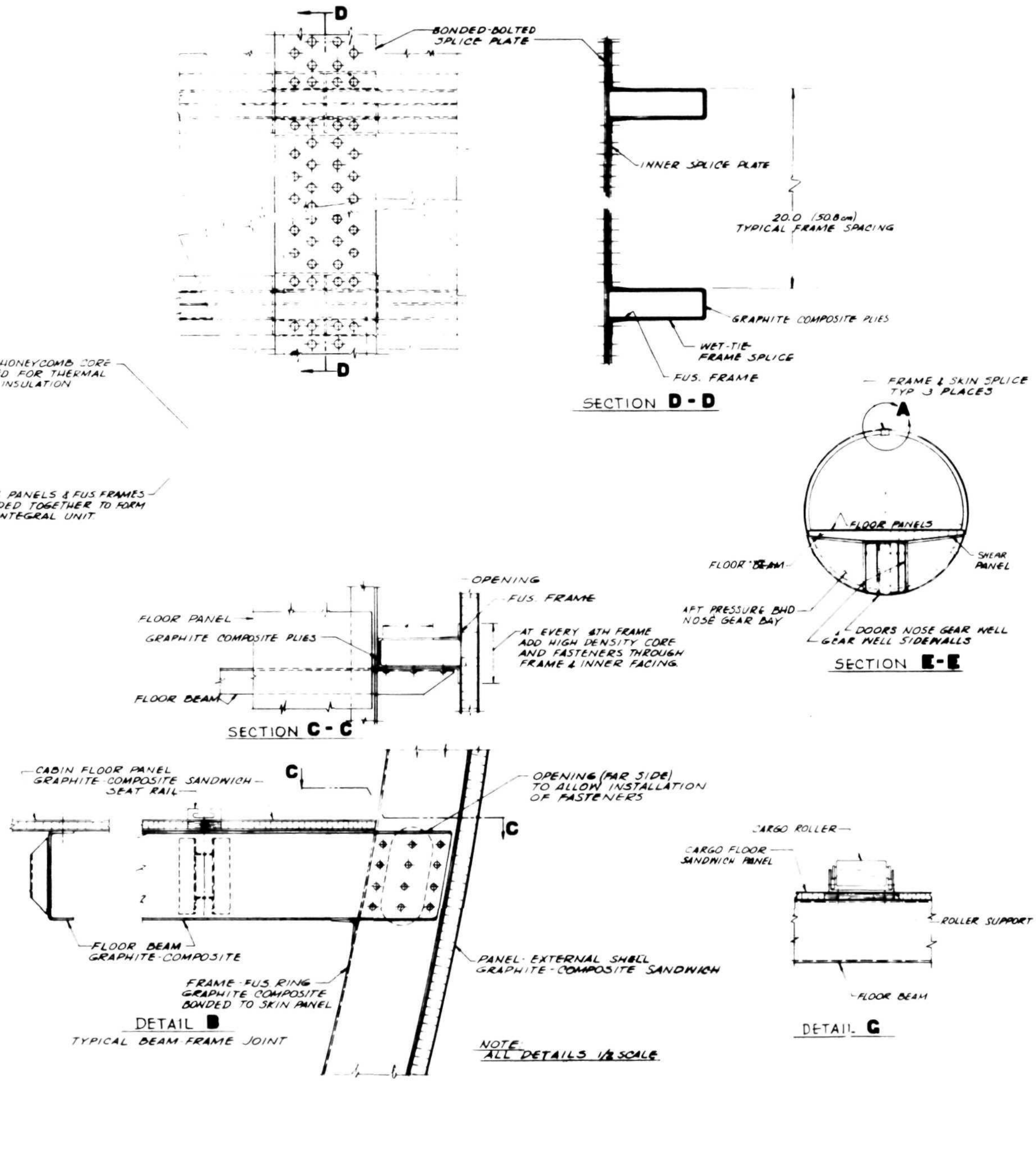


Figure 3.1-12 Frame Splice

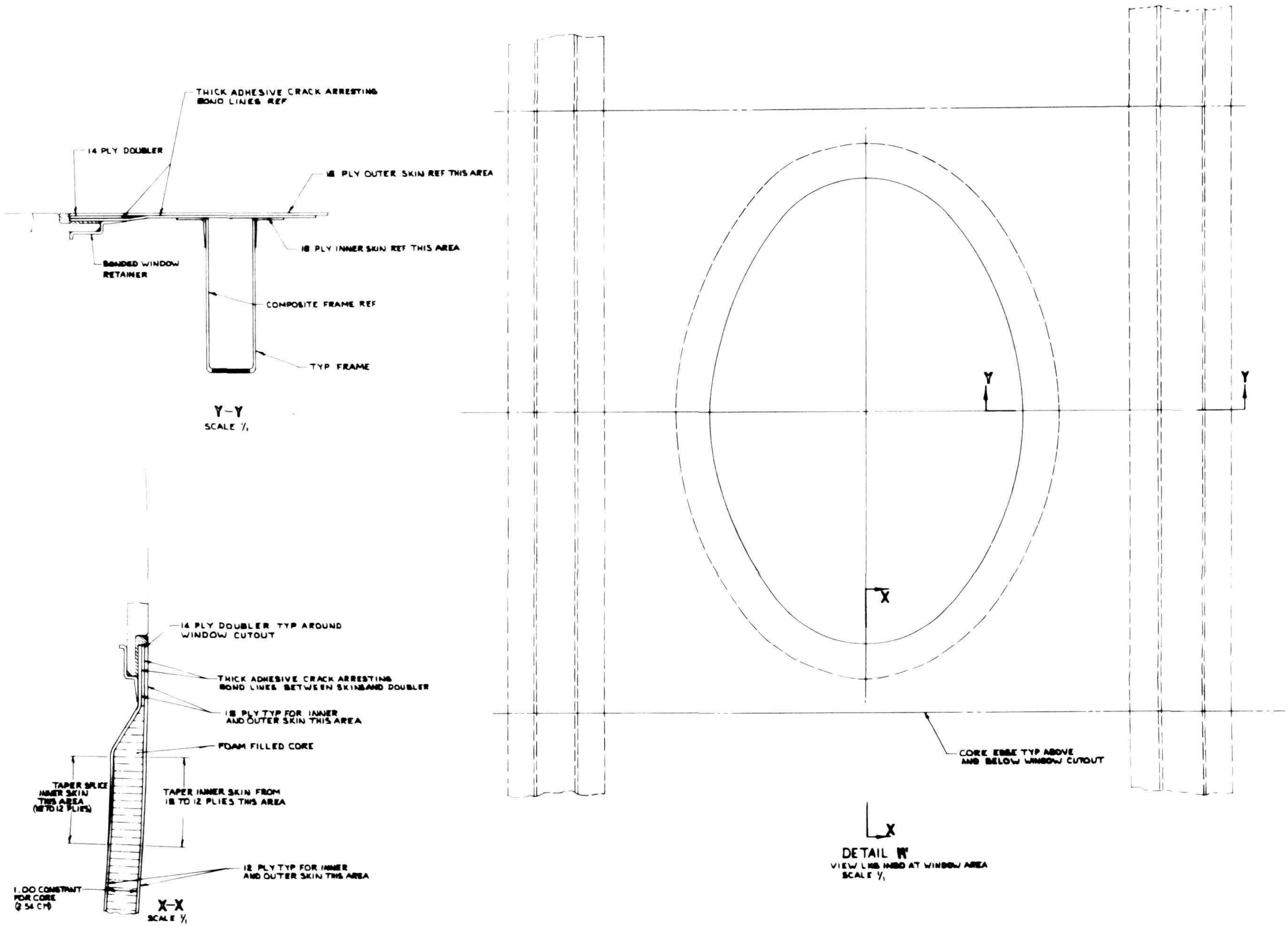


Figure 3.1-13 Fuselage Window

Internal pressure bulkheads are located beneath the cabin floor at each end of the nose wheel bay and at the aft end of the main wheel bay as shown in Figure 3.1-14. At the forward end of the main wheel bay and wing cutout, the front wing spar web serves as the pressure barrier. These bulkheads are of sandwich design using graphite-composite facings and aluminum core. Hat section fiberglass and graphite-composite stiffeners are bonded to the bulkhead for pressure load transfer above each landing gear well and in the over-wing area. The sandwich floor panels also serve as pressure seal webs. Longitudinal shear webs forming the sides of the nose wheel bay as shown in Figure 3.1-14 separate the landing gear from the electrical system equipment and transfer the gear loads to the pressure bulkheads and skins. Between the main landing gear wheels running longitudinal are shear webs which transfer landing gear door loads and also provide a path for bending loads from the aft fuselage to the wing. These shear webs are fabricated of graphite-composite sandwich stiffened with graphite-composite hat sections bonded to the webs.

The aft pressure bulkhead is a membrane type graphite-composite web as shown in Figure 3.1-15. It is installed as a single piece on final assembly.

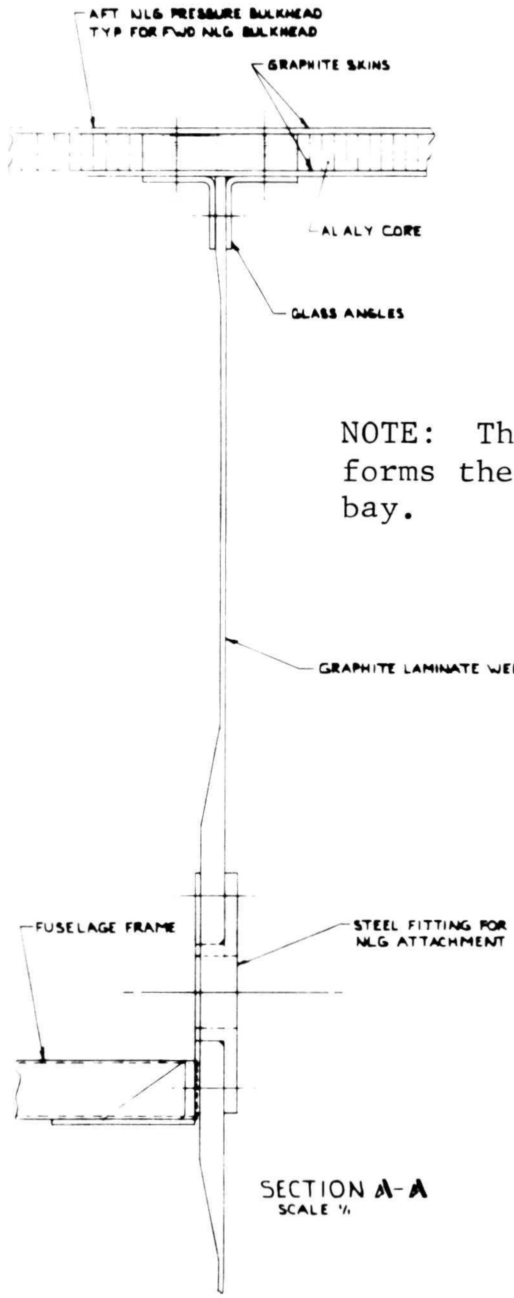
3.1.3.3 Door Structure

The passenger doors are conventional type with graphite-composite skins over aluminum honeycomb core and operate on metal tracks. The doors are of the inward opening retracting plug type. The door plug and the fuselage skin around the door opening are reinforced with graphite laminate as shown in Figure 3.1-16.

Cargo doors are conventional in design and are hinged along the upper surface and open outwards. Latches along the lower surface provide hoop tension continuity when the doors are closed and locked. The hoop tension door design, through its good structural continuity of the fuselage, results in a minimum weight, reliable installation. The doors are shown in Figure 3.1-17.

3.1.3.4 Wing-Fuselage Attachment

Wing to fuselage attachment is accomplished through three primary fuselage frames and drag ties along the upper and lower wing skin surface. The fuselage frames are laminated



NOTE: The graphite laminate web forms the side of the nose wheel bay.

Figure 3.1-14 Nose Wheel Bay

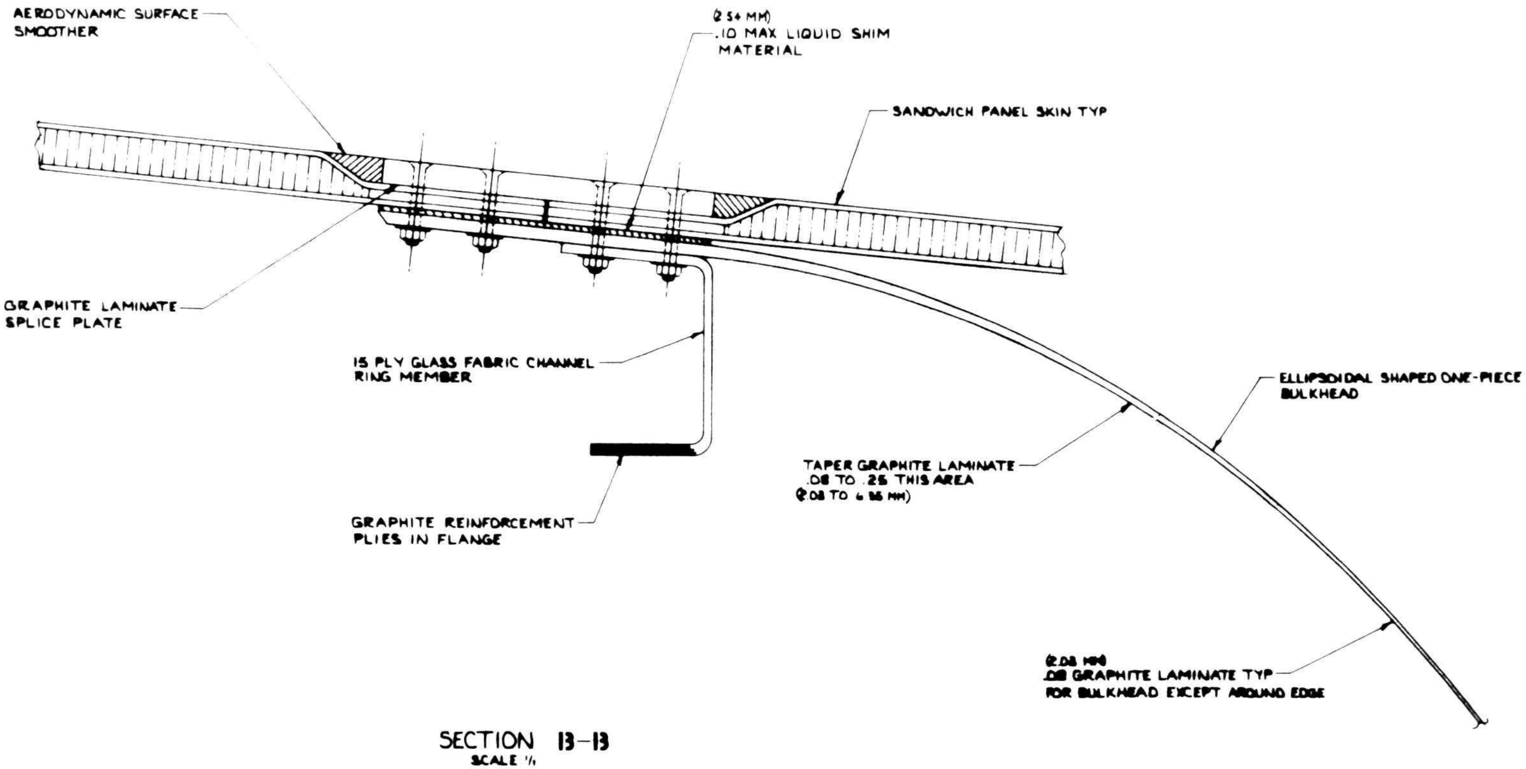
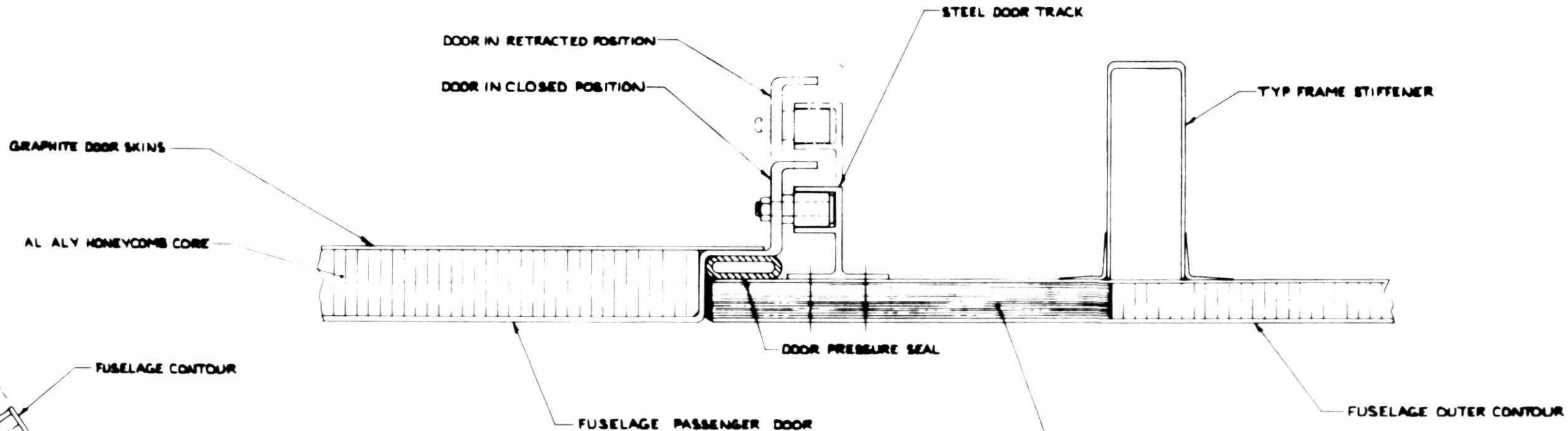
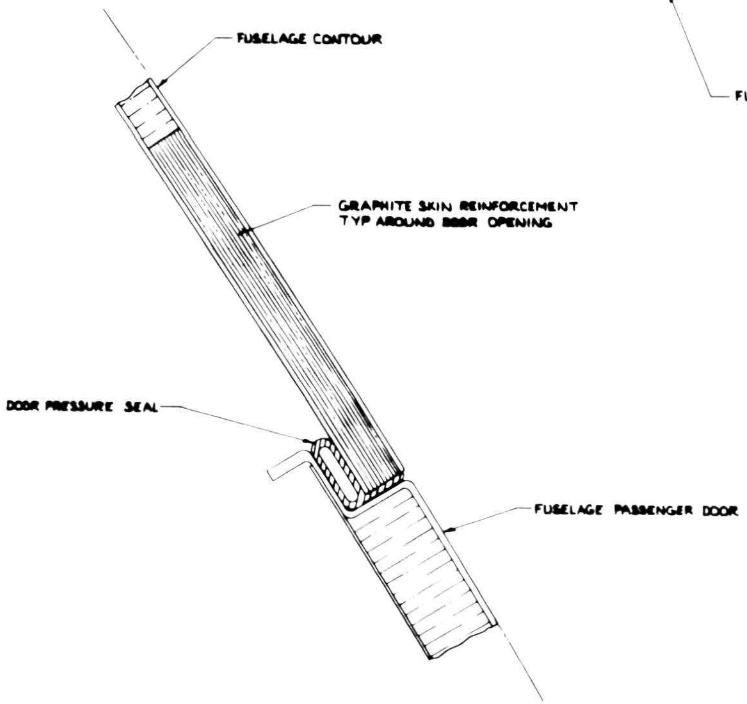


Figure 3.1-15 Aft Pressure Bulkhead

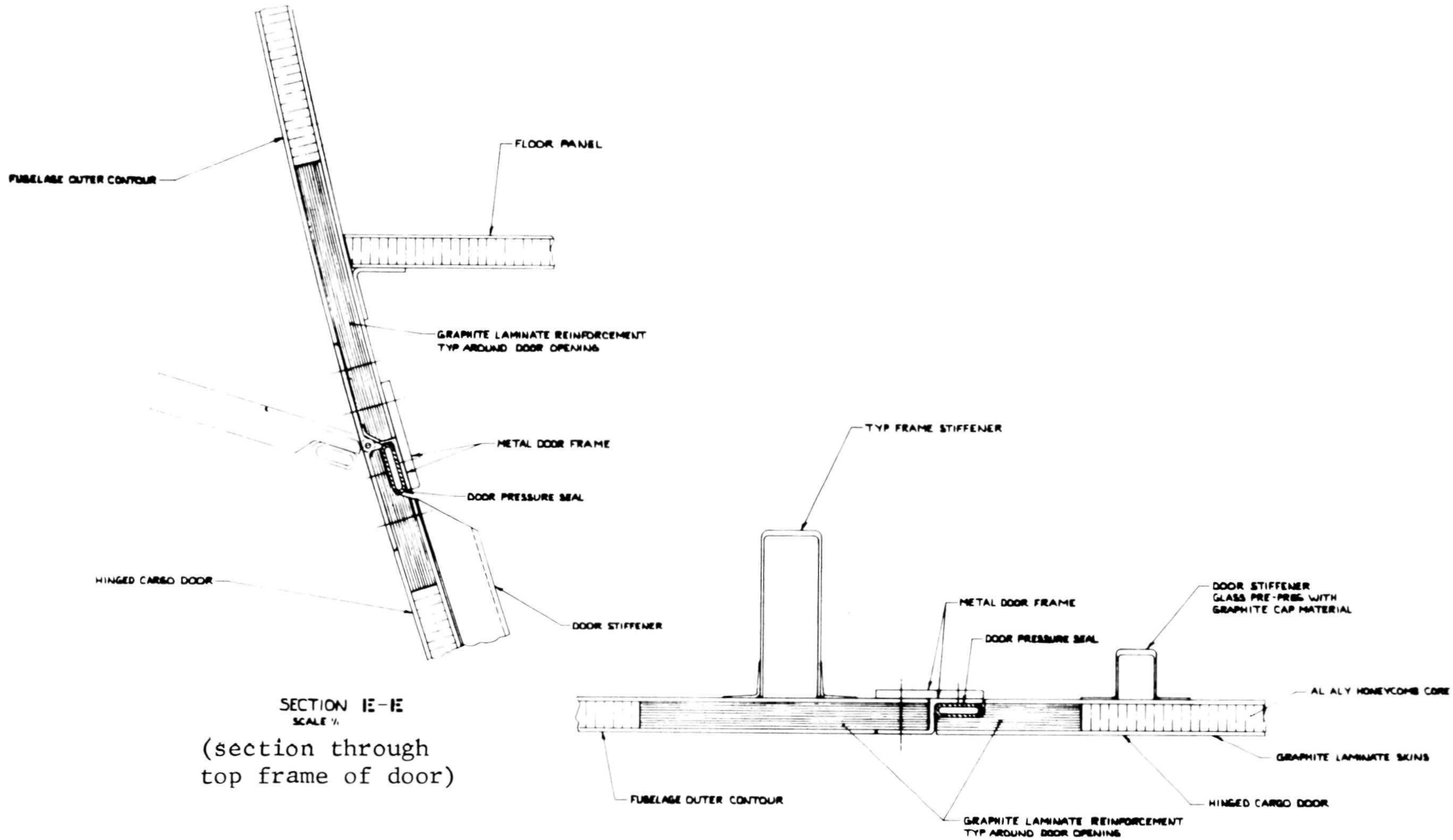


SECTION A-A
(section through side frame of door)



SECTION B-B
(section through top frame of door)

Figure 3.1-16 Passenger Door



SECTION E-E
SCALE 1/4
(section through top frame of door)

SECTION D-D
SCALE 1/4
(section through side frame of door)

Figure 3.1-17 Cargo Door

graphite-composite with concentrated build-ups of directionally oriented 0 degree graphite reinforcing plies, as compared to the metal frames reported in Reference 2-1. The fuselage attaches to the front and rear wing box spars as shown in Figure 3.1-3.

3.1.4 Horizontal Tail

The horizontal tail is a one-piece all movable tail-plane hinged at the aft edge of the center box and actuated by four hydraulic actuators mounted two above and two below the box. The horizontal tail structure consists of the primary structural box and secondary structure composed of the leading edge, trailing edge, and tip. The horizontal tail structure is shown in Figure 3.1-18.

3.1.4.1 Primary Structural Box

The primary structural box is continuous through the aft fuselage section and has a splice located on the airplane centerline. The box has a front spar at the 37.0 percent chord line, a rear spar at the 61.5 percent chord line, upper and lower skins and multiple ribs. The front and rear spars are fabricated of sandwich construction with graphite-composite facings and aluminum honeycomb core. The upper and lower skins are also sandwich construction of graphite-composite laminated facings over aluminum honeycomb core. The facing laminate orientation and thickness are tailored to provide optimum load capacity. The box ribs are of laminated graphite-composite with bonded fiberglass stiffening. The lower skins and spars are attached by nonexpanding shank rivets and the ribs are bonded in. The removeable upper skin is mechanically attached to the box.

3.1.4.2 Secondary Structure

Horizontal tail secondary structure consists generally of graphite-composite laminated skin panels using fiberglass stiffening. Intermediate spars are located on the 8.0 percent and 80.0 percent chord lines. Leading edge segments and tips are interchangeable and are easily removed and replaced.

3.1.4.3 Support and Actuation

The horizontal tail is supported at two locations, as shown in Figure 3.1-19, each by double clevis joints. The single lug of each joint is attached to the tail structural

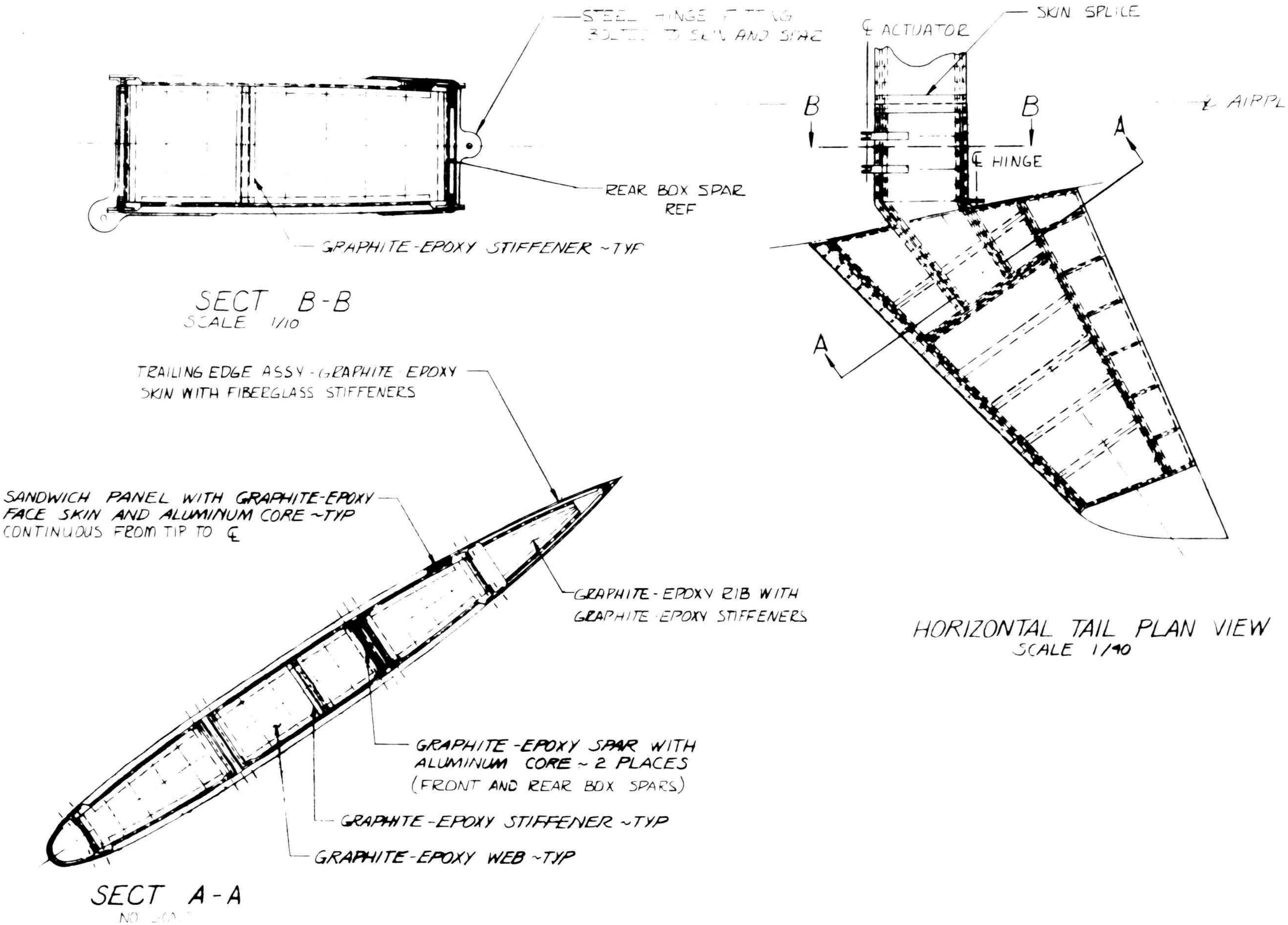


Figure 3.1-18 Horizontal Tail Structure

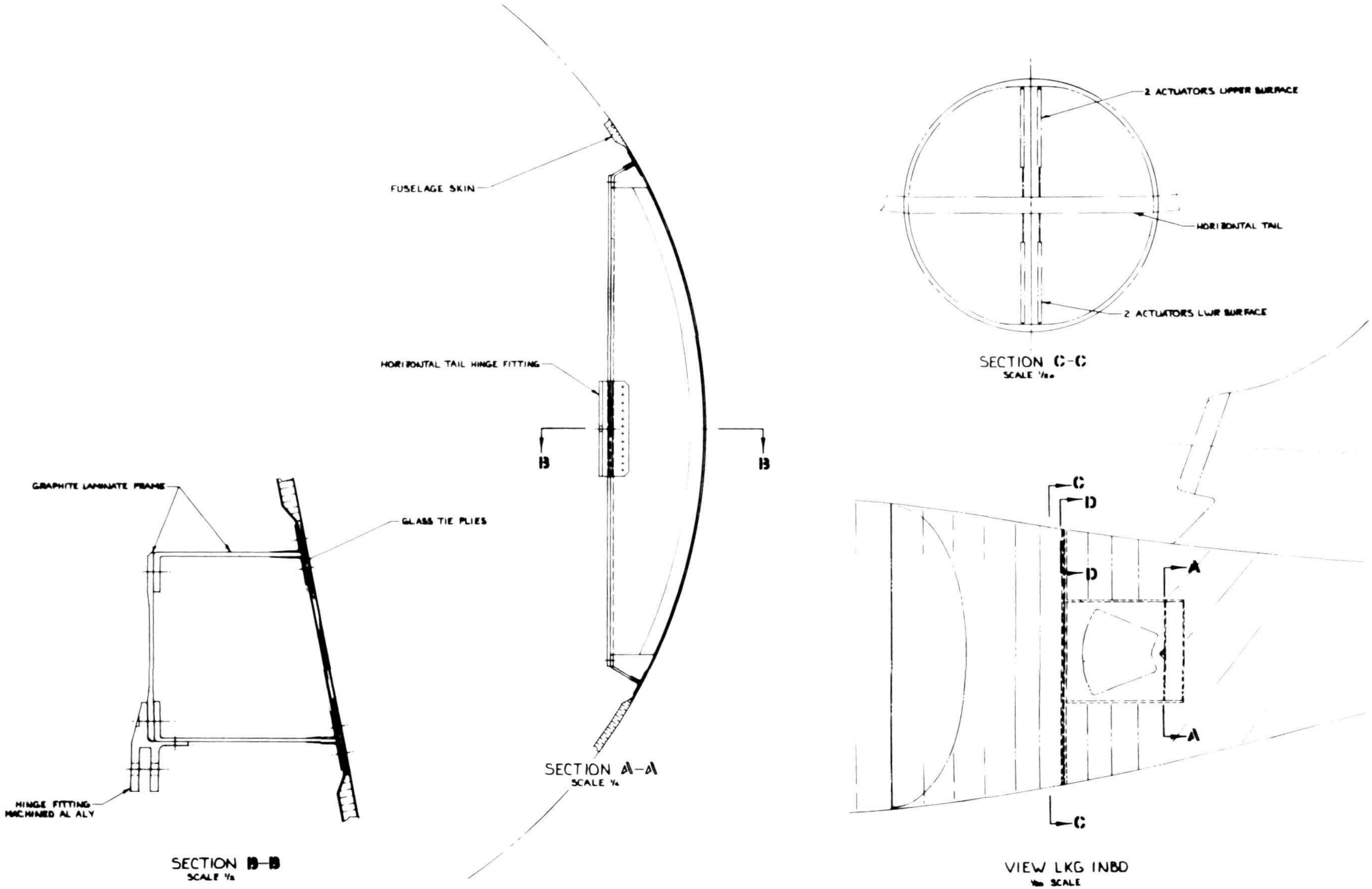


Figure 3.1-19 Horizontal Tail Support

box while the double lug fitting is attached to the fuselage bulkhead. The single lug contains the bearing insert and is composed of two laminates of metal for fail safety. A rib is located in the structural box forward of and in line with the support fitting.

The actuator fittings are forged aluminum or steel fittings bolted to the upper surface of the center box. As in the hinge fittings, the actuator fittings are backed up by a center box rib. The redundant actuators are sized and located for fail safety.

3.1.5 Vertical Tail

The vertical tail structure consists of a fixed multi-spar primary box supporting both a rudder and a tail mounted engine. Fixed secondary structure in the leading edge, tip, and trailing edge access panels complete the vertical tail structure. The structural arrangement of the vertical tail may be seen in Figure 3.1-20.

3.1.5.1 Primary Structural Box

The primary structural box is a four-spar box between the 15.0 percent and 55.0 percent chord lines. The spars are one piece full length components from the top of the vertical tail to their termination point down in the aft fuselage. They split to form rings around the engine inlet area as shown in Figure 3.1-21. There are four spars for added fail safety. The box covers above the nacelle are sandwich skin panels that pick-up air loads and provide bending material for the structural box. In the area around the nacelle and down to the top fuselage line, the bending loads are carried in directionally oriented cap material located in the extremities of the spar web. Ribs are located to support the spars and rudder hinge points as shown in Figure 3.1-22.

The primary box spars are formed of graphite composite webs separated by aluminum honeycomb core in the area above the nacelle and below the upper fuselage contour. The core is omitted in the ring members around the nacelle and in the spar web between the nacelle and the upper fuselage contour to provide space for the web cap material. The box sandwich skins are fabricated of graphite composite facings over aluminum honeycomb core as shown in Figure 3.1-23. The core is omitted in the area of the spars to facilitate attachment.

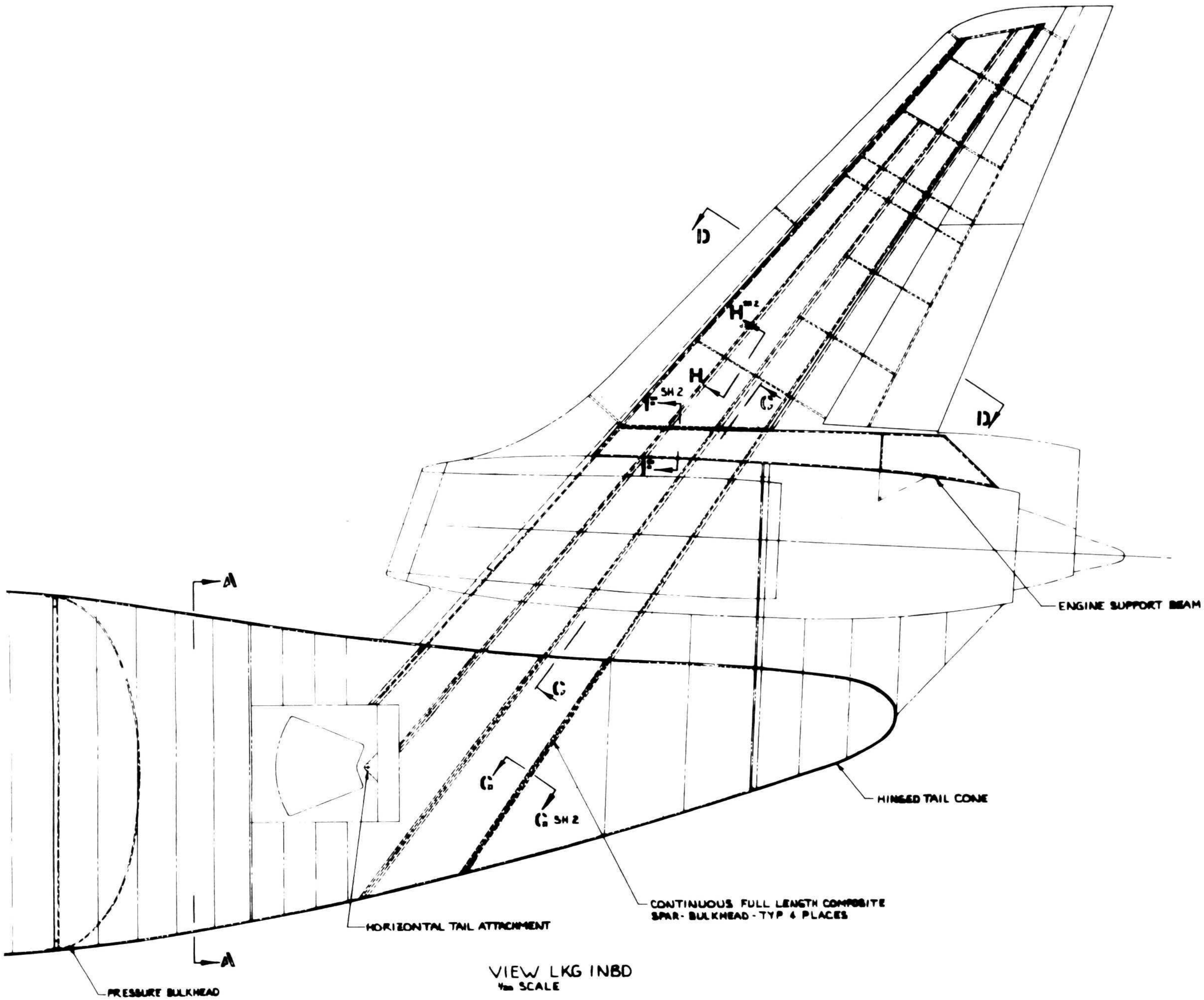


Figure 3.1-20 Vertical Tail Structural Arrangement

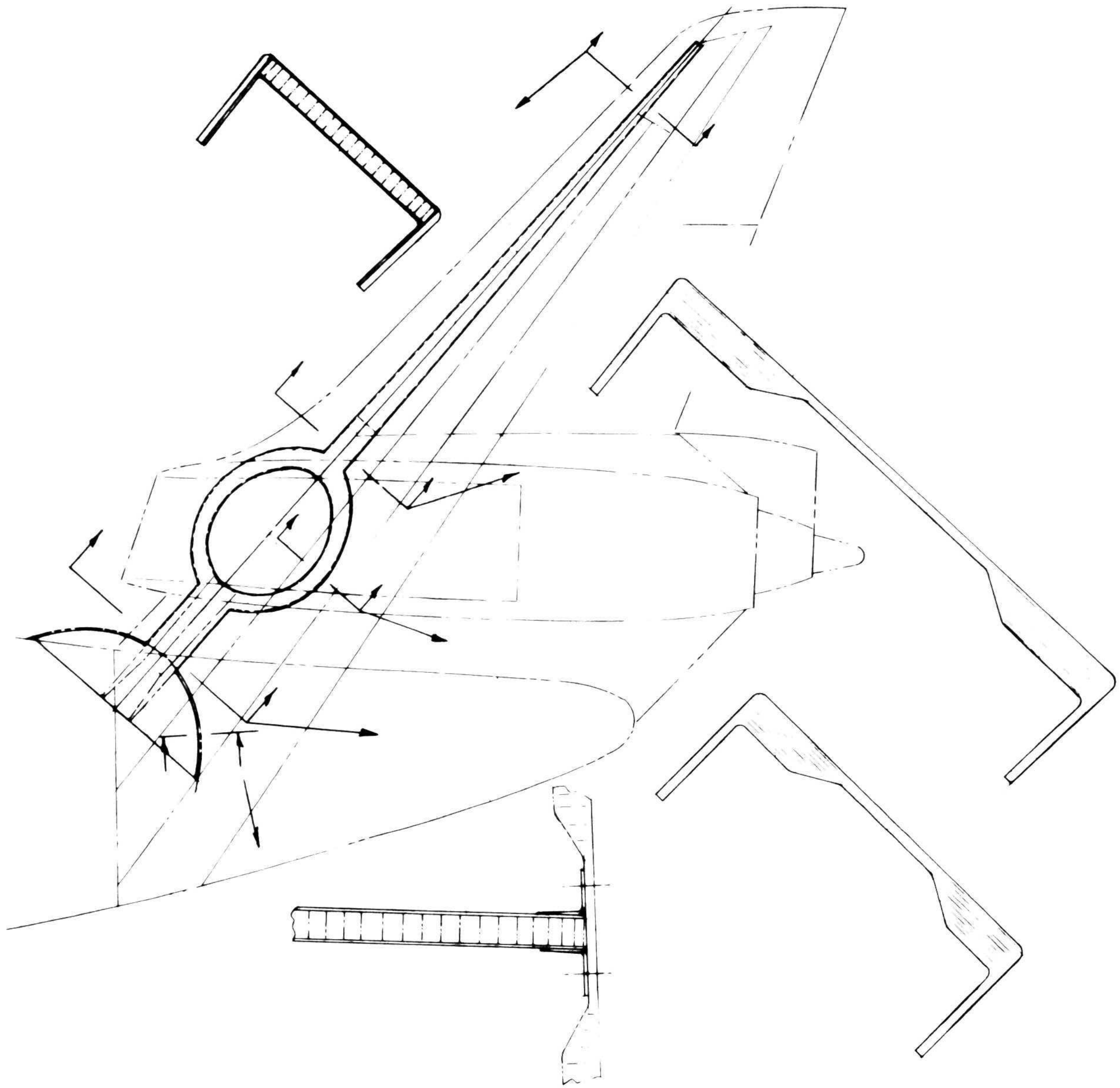


Figure 3.1-21 Vertical Tail Spar

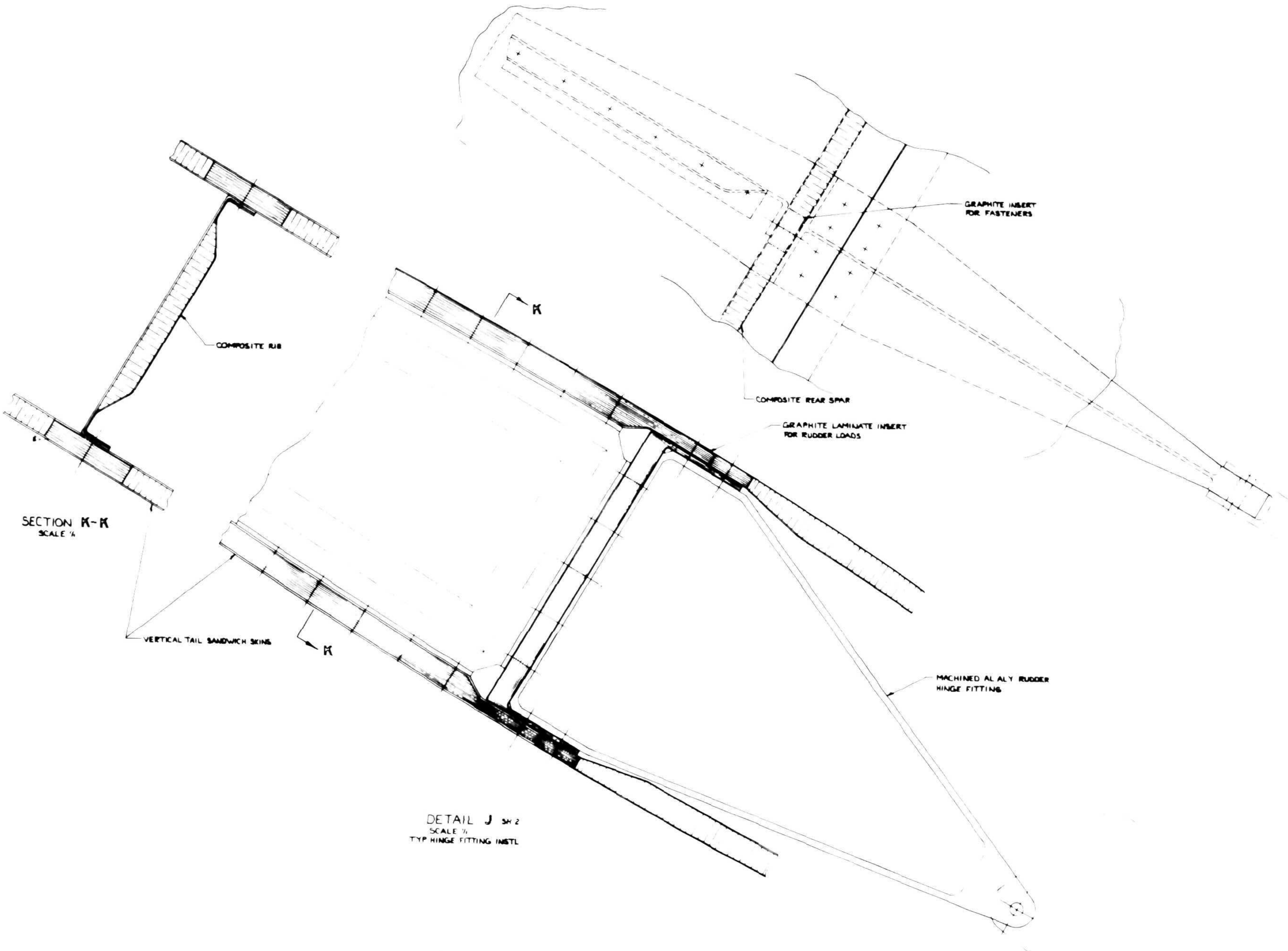
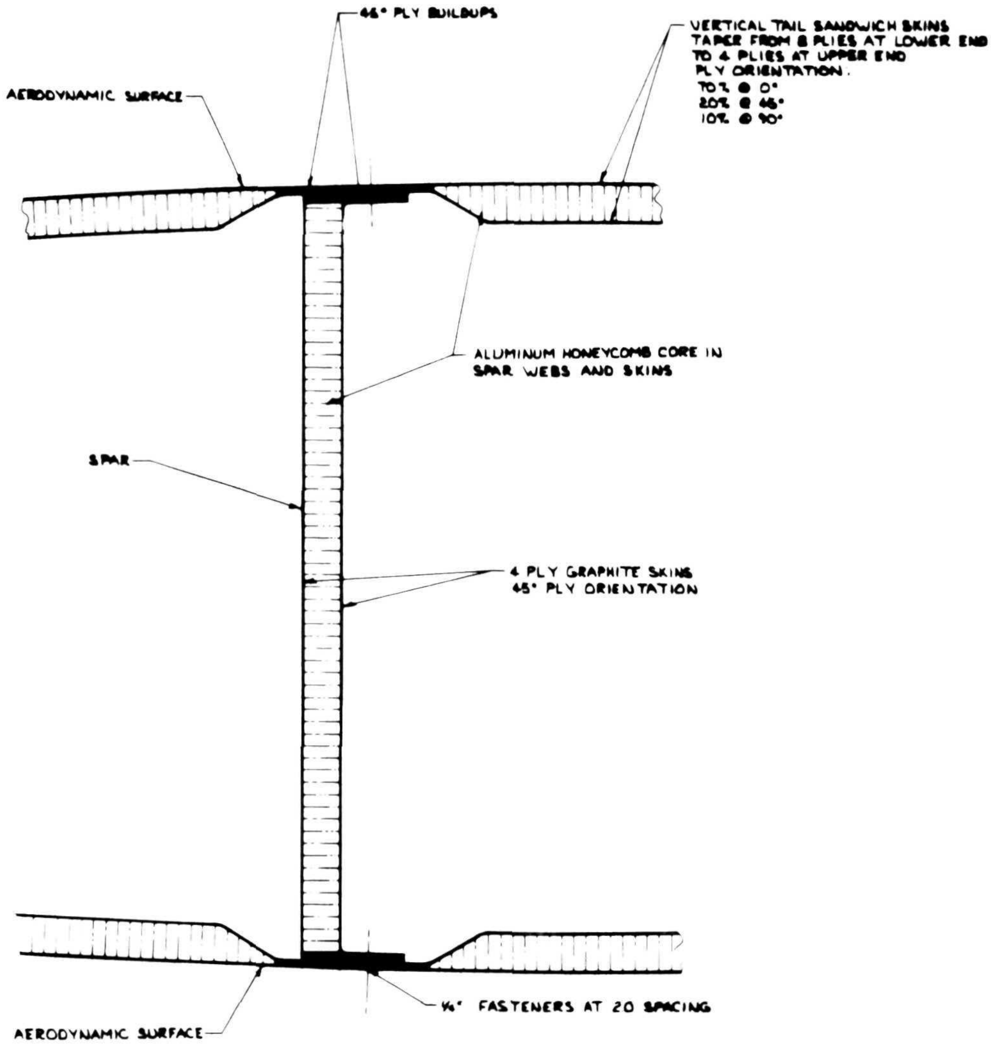
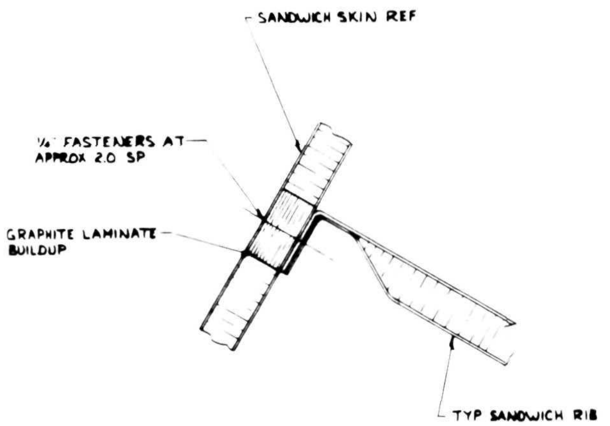


Figure 3.1-22 Rib and Rudder Fitting Attachment



DETAIL E
FULL SCALE



SECTION H-H
TYP VERTICAL TAIL SKIN TO RIB ATTACHMENT
FULL SCALE

Figure 3.1-23 Vertical Tail Skins

A full depth graphite-composite spacer is installed at the rib attachment to provide a smooth continuous spanwise inner box skin surface as shown in Figure 3.1-23. Rib segments are formed of graphite webs over aluminum honeycomb core similar to the upper spar sections.

The engine support is a cantilever beam that attaches to the spars just above the nacelle ring area as shown in Figure 3.1-24. The beam consists of a graphite laminate web with concentrations of directionally oriented 0 degree fibers to provide bending strength. The external beam surface forms vertical tail external contour. Mechanical fasteners are located in areas where a strain tolerant graphite laminate is provided to reduce stress concentrations.

3.1.5.2 <u>Secondary Structure</u>

The rudder consists of two sections, the upper and the lower, from the 68 percent chord line aft. Each rudder half can be operated independently for fail safety. The rudder's fiberglass-composite substructure consists of a single spar with chordwise formed ribs terminating on the aft closing channel. The skins are bonded sandwich panels with graphite-composite facings, aluminum core and fiberglass web layup edge members at the substructure intersection. The rudder segments are supported from the aft spar by forged aluminum hinge fittings at four places for each rudder half. Access doors between the rear spar and rudder are hinged from the rear spar for quick easy operation. These doors are sandwich panel construction using graphite composite facings and aluminum honeycomb core.

The fixed leading edge segments and tip are of a similar construction to the wing and horizontal tail leading edge and tip structure. These removable, replaceable segments have chordwise formed ribs covered by bonded sandwich skins. The segments are assembled by bonding and riveting and installed using mechanical fasteners.

3.1.6 Nacelle and Pylon

The engine nacelles consist of two wing mounted units and one unit mounted in the vertical tail. Each nacelle is made up of a nose cowl assembly, fan cowl, and fan duct cowl assembly. The wing mounted engines are supported by a pylon cantilevered forward and below the wing while the tail mounted engine is supported by a beam extending aft of the vertical

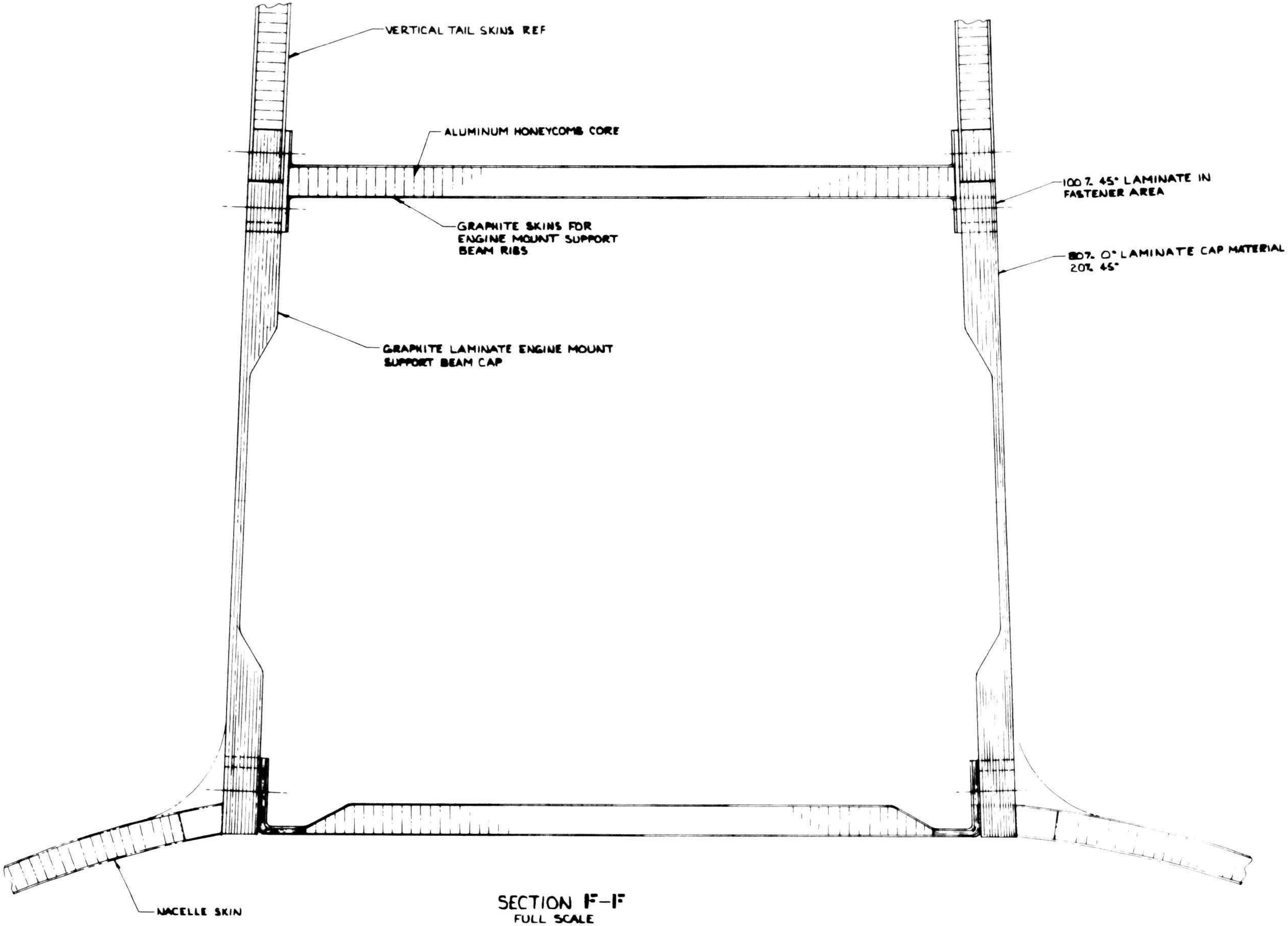


Figure 3.1-24 Engine Support Beam

tail structural box. The mechanical nacelle arrangement is shown in Figure 3.1-25.

3.1.6.1 Nose Cowl Structure

The nose cowl consists of a translating leading edge fixed cowl, inlet duct, built-in pylon disconnect, and the noise treatment secondary structure as shown in Figure 3.1-26. The leading edge structure consists of graphite-composite inner and outer skins and radial frames of laminated sheet. The forward portion of the pylon is integrally built into the fixed cowl to provide adequate structure to support the cowl from the pylon and disconnect. The fixed cowl and inlet consists of graphite-composite formed frames and aluminum machines frames, machined longerons, and graphite-composite sandwich inner and outer skin panels.

The inlet duct wall surface is acoustically treated with honeycomb material, covered with a perforated graphite-composite skin. Further noise attenuation is provided by two concentric splitters that are 1 in. (.025 m) thick for a length of 36 in. (.91 m). These splitters are supported by three radial vanes and are constructed of honeycomb material and covered with perforated graphite-composite skin.

3.1.6.2 Fan Duct Cowl Structure

The fan duct cowl structure consists of two integrated doors, one on each side of the engine, that contain the thrust reverser, the outer structure, the fan duct, and noise attenuation treatment. These doors, when opened, expose the entire engine and the engine/airplane accessories. The doors are constructed of half round frames and longerons and skin panels. The inner structure consists of a fan duct, noise treated surfaces, and splitters. The bottom portion has a smooth surface which houses the engine/airplane accessories.

Door framing and supporting structure are constructed of aluminum. The core panels must withstand the high temperature of the engine, therefore, extra thickness graphite skins are used. The facing on the fan duct side is perforated for acoustic treatment. The splitters and external fan duct skins are sandwich with perforated graphite skin on the duct surfaces and graphite-composite structural skins. Outside sandwich skin panels are graphite-composite facings with aluminum honeycomb core.

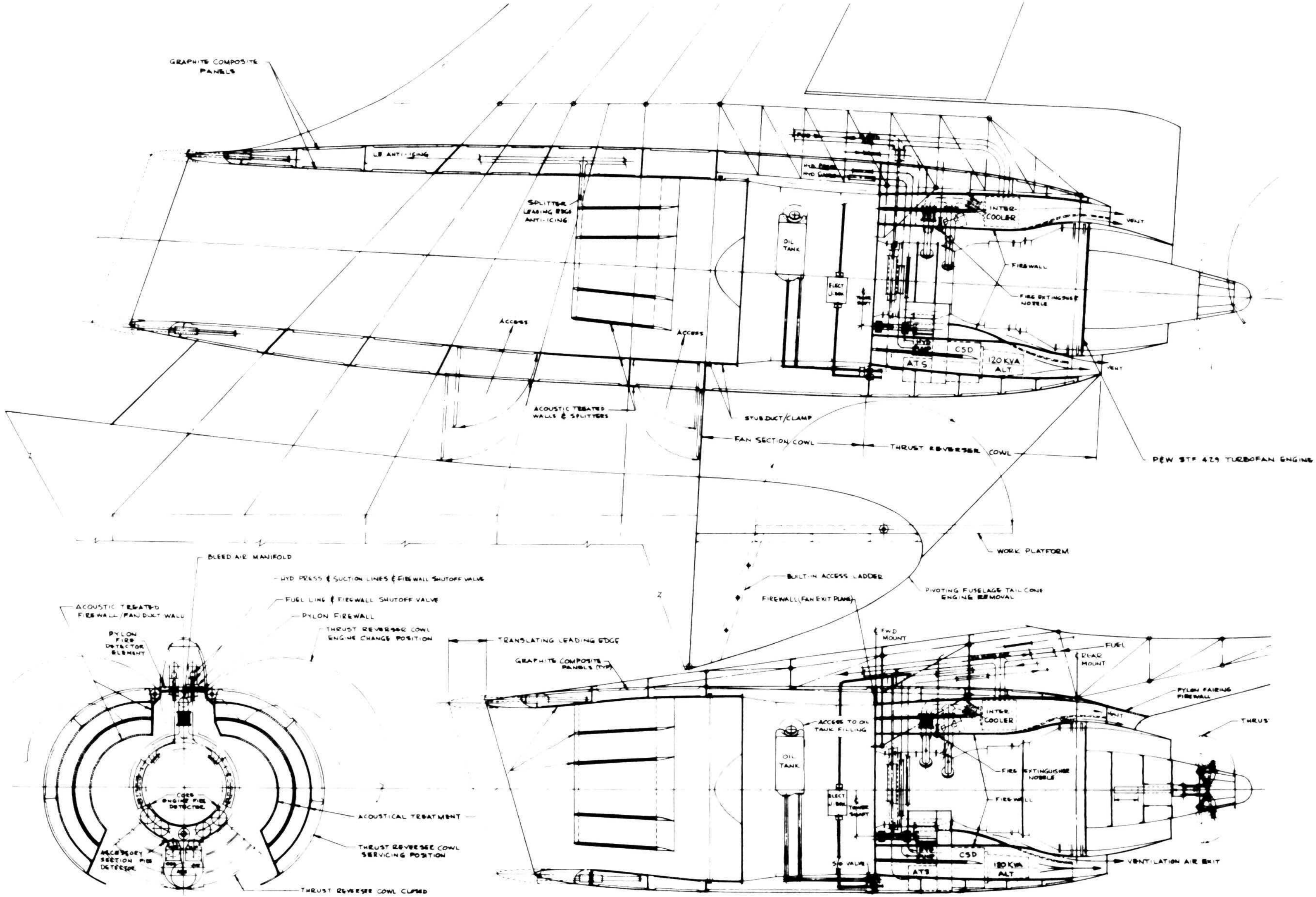


Figure 3.1-25 Nacelle Mechanical Arrangement

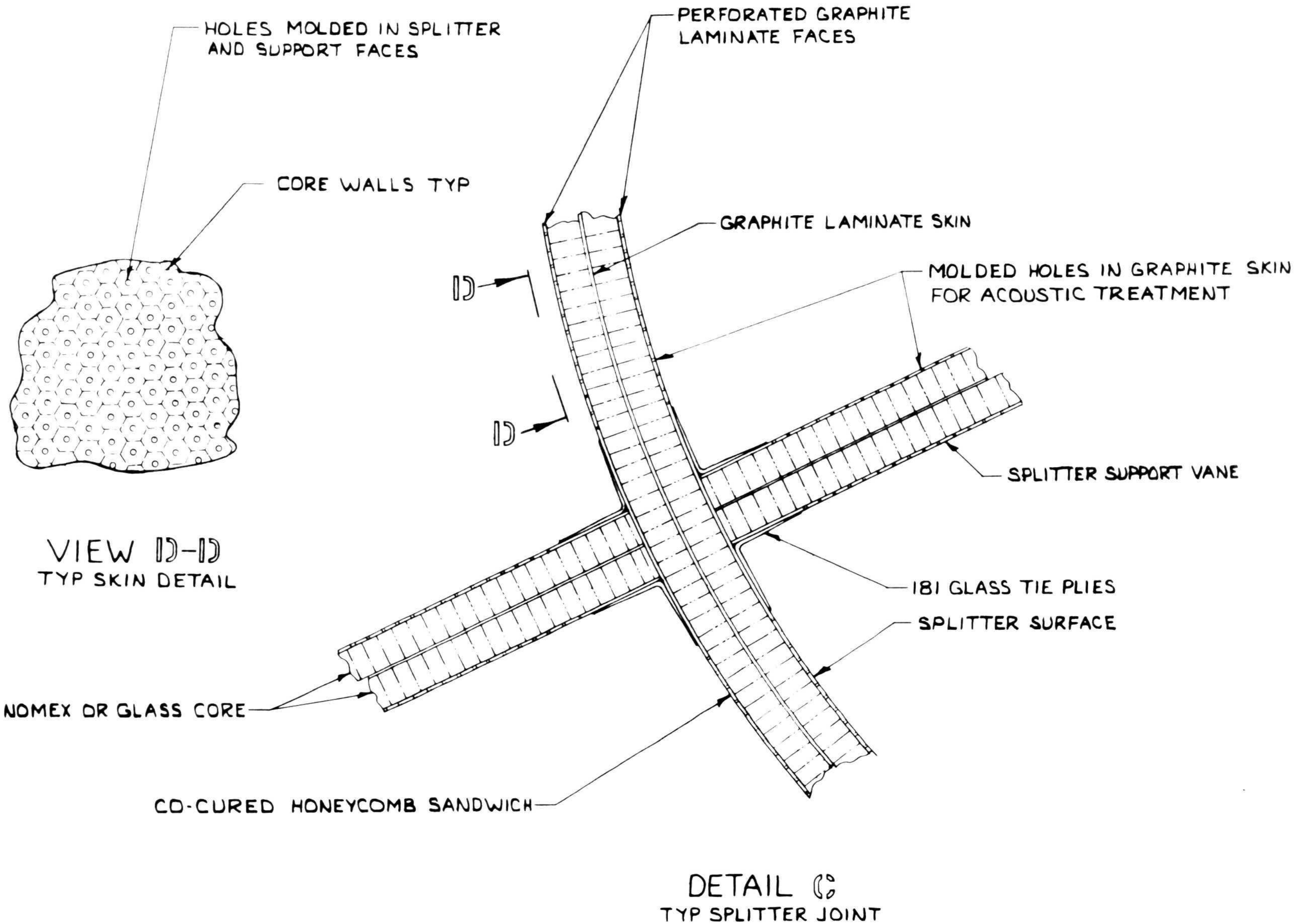


Figure 3.1-26 Nacelle Noise Treatment

3.1.6.3 Pylon Structure

The wing mounted pylon structure consists of a welded steel frame as shown in Figure 3.1-27. The attachment to the wing box is accomplished by attaching to the upper and lower wing skins in the area of the front spar only, eliminating the necessity for cutouts in the lower sandwich panel wing skin. The fairing material over the welded frame is graphite-composite laminated skin with fiberglass stiffening ribs and attachment angles.

3.1.7 Alternate Structural Concepts

The primary method of construction employed in both the wing and fuselage sections is the co-curing of graphite-composite honeycomb sandwich. The method of construction was exploited primarily because of its ability to be fabricated economically in large sections utilizing fewer detail parts and major assembly fixtures. Additional studies were conducted to investigate alternate economic approaches to wing and fuselage construction.

3.1.7.1 Wing Concepts

Figure 3.1-28 shows a shell-liner approach to wing fabrication in which the wing box skin is stabilized with spanwise stiffeners in place of the continuous honeycomb core. In the chordwise direction, composite ribs with bonded attachments are used to break up the panel size and to introduce leading and trailing edge loads. Essentially both the upper and lower coverings are formed of continuous inner and outer skin facings. The outer skin facing is the mold surface and contains the chordwise material for chordwise bending loads and bias oriented material for torsional rigidity. The inner skin forms the spanwise stiffeners and is composed of spanwise oriented material for wing bending loads with a small amount of biased woven graphite cloth for formability.

3.1.7.2 Fuselage Concepts

The fuselage shell liner concept is shown in Figure 3.1-29. The main portion of the fuselage is constructed in three large segments as in the honeycomb sandwich approach. The skin is stabilized with longitudinal hat section stiffeners in place of honeycomb core. Conventionally spaced ring members are used for overall shell stability. The shell consists of continuous inner and outer graphite laminated

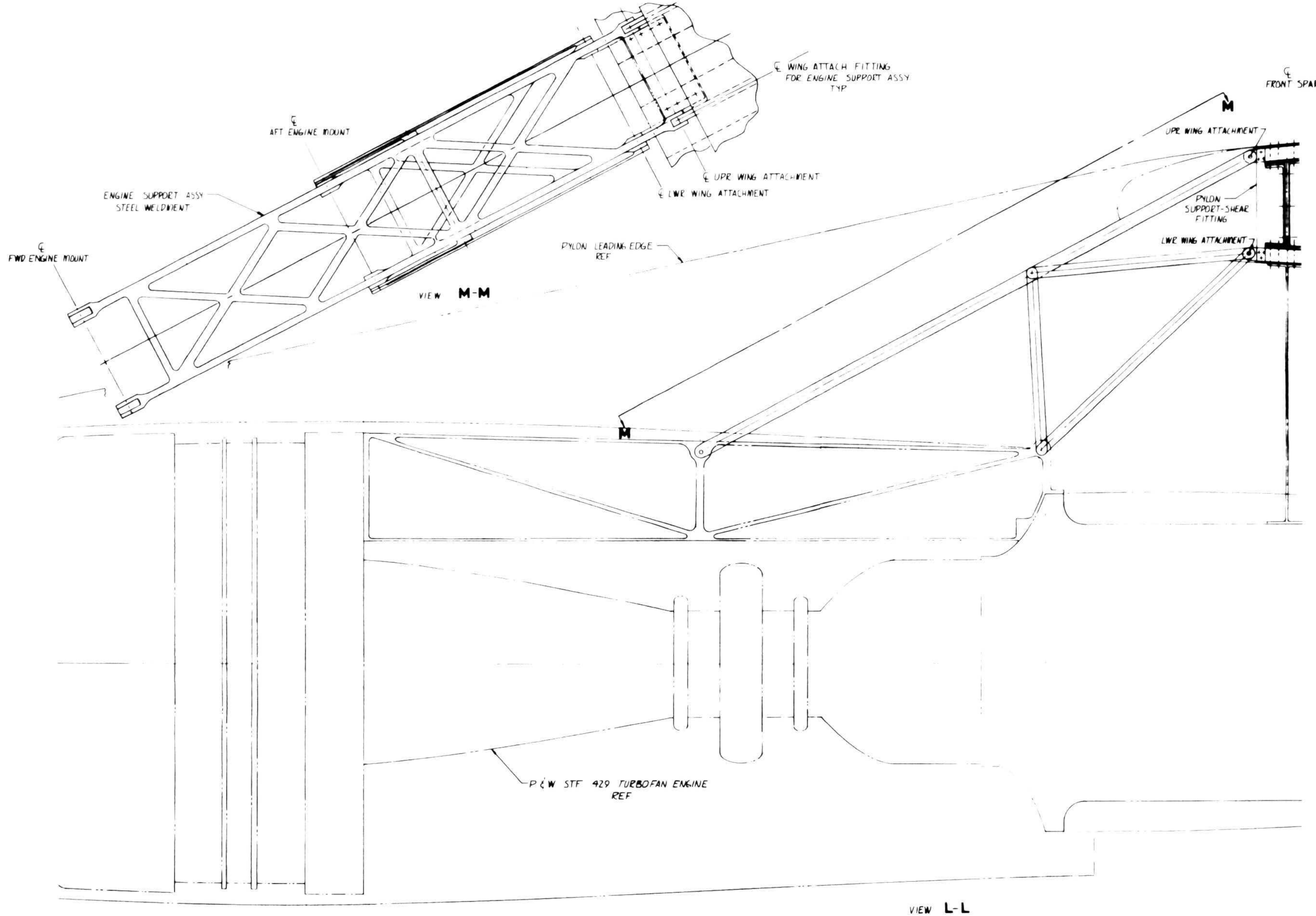


Figure 3.1-27 Wing Pylon Structure

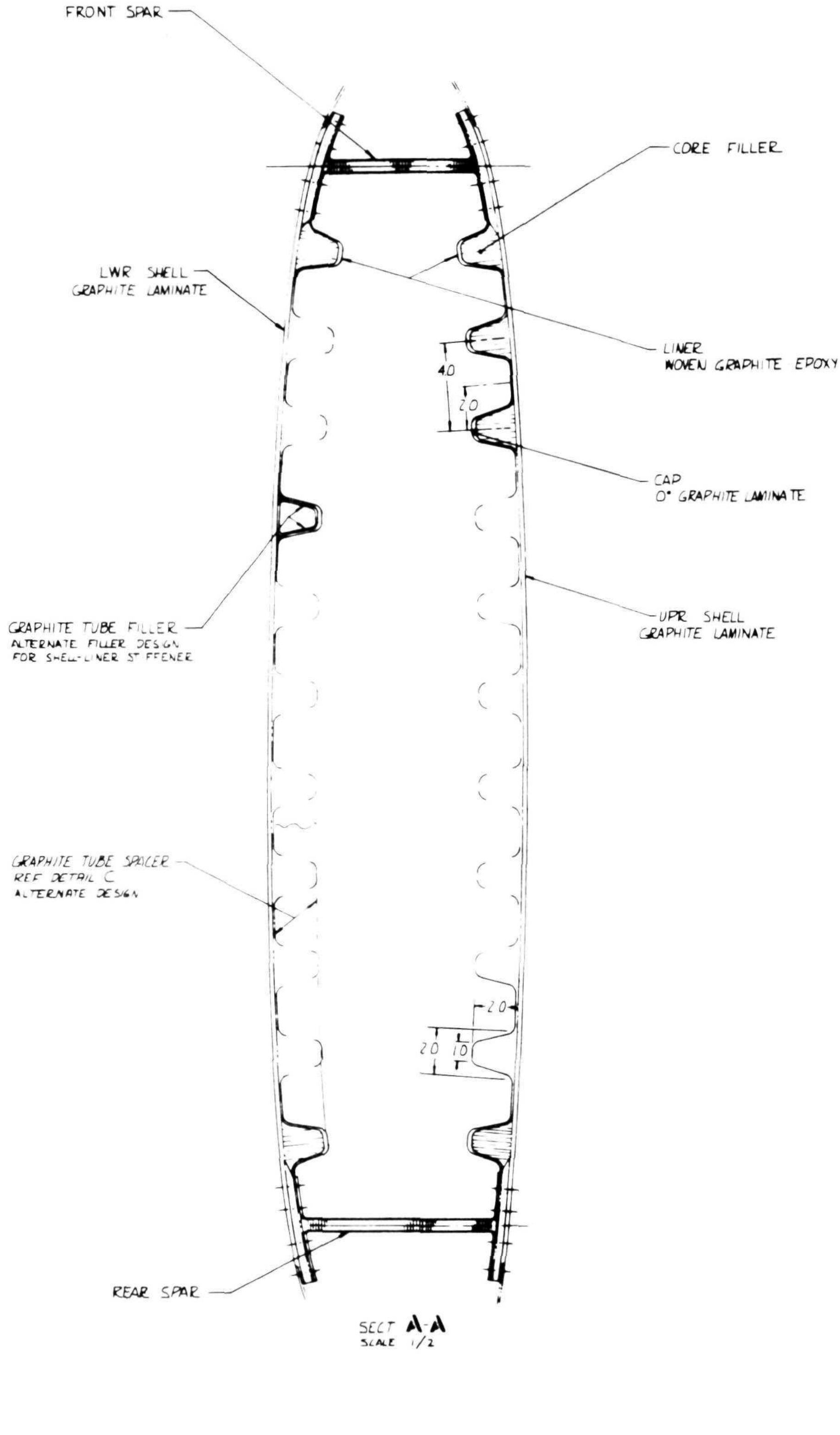


Figure 3.1-28 Wing Structural Concepts

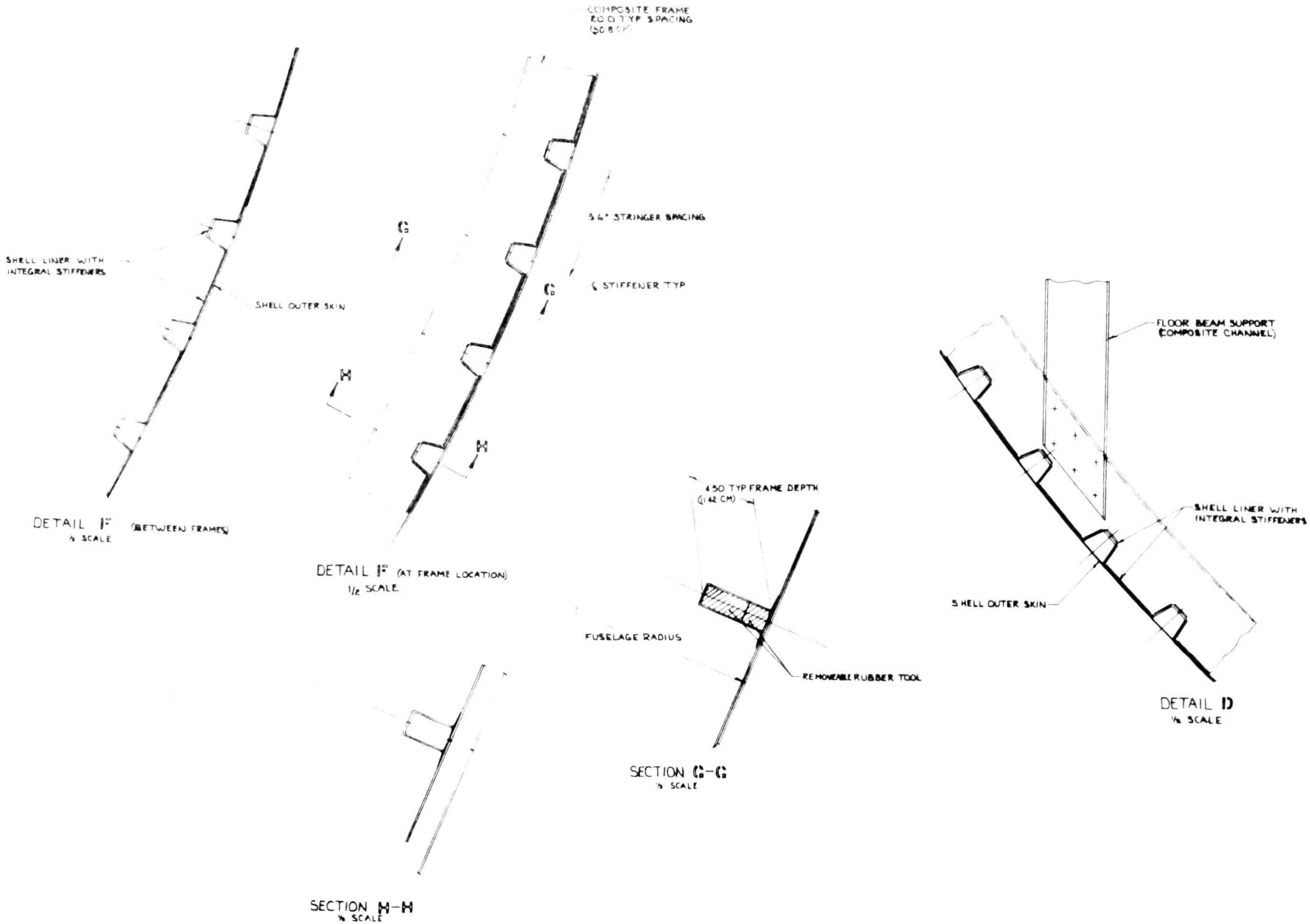


Figure 3.1-29 Fuselage Structural Concepts

skins. The outer skin is the mold surface and is laminated of circumferentially oriented plies to carry the hoop tension loads and biased plies to carry the body shear loads. The inner skin forms the hat section stiffeners over removable rubber tooling and is laminated with longitudinally oriented plies to carry body bending loads and biased woven graphite cloth for formability. The ring stiffeners are bonded in place, also over removable rubber tooling.

3.1.8 Thermal and Acoustic effects

Three fuselage skin constructions were considered for the transport; namely (1) an aluminum sheet with aluminum stiffeners, (2) a graphite-composite sheet with graphite-composite stiffeners, and (3) a 1 in. (.025 m) sandwich panel consisting of graphite-composite skins and an aluminum honeycomb core. In the overall evaluation of structurally equivalent skin constructions, comparisons are made of the effectiveness of the above panels as thermal insulators and attenuators of acoustic noise.

If these panels are considered by themselves and not how they interact with the primary insulation in the fuselage walls and the environmental control system, the sandwich panel provides the best insulating characteristics with the laminated graphite-composite sheet second and the aluminum last. However, these skins should not be evaluated as independent elements but should be considered as a part of the overall fuselage thermal insulation system which consists of the skins, the frames, the primary thermal (and acoustical) insulation and interior wall. Considering the overall insulation system (skin, frames, primary insulation and interior wall), simple calculations show that the thermal resistance of even the honeycomb sandwich panel is very small compared with a reasonable thickness of primary insulation. Alternately, if a one-inch-thick (.025 m) primary insulation is required for the aluminum skin construction, the same thermal insulation can be provided by the one-inch (.025 m) sandwich panel and about .092 inches (.023 m) of insulation. Therefore, because a primary thermal insulation is required, the skin construction will contribute little to the overall thermal insulation of the fuselage wall. Also, the capacity of the environmental control system is influenced to a negligible extent by the heat transfer through the fuselage walls, since its design is based primarily on the heating or cooling of the large amount of air required for passenger and crew compartment ventilation.

The acoustic noise transmission loss (T_L) of the fuselage panels has been predicted for the three types of construction and the results are shown in Figure 3.1-30.

Some tentative conclusions which can be drawn from these predictions are:

1. In the frequency range below the fundamental resonance, the honeycomb panel affords significantly higher T_L because of its greater stiffness and higher resonant frequency.

2. In the frequency range between the fundamental resonance and the coincidence dip, mass is the controlling parameter and the aluminum plate stringer construction has superior T_L.

   NOTE: The coincidence frequency, f_c, is the frequency at which the local velocity of sound coincides with the velocity of propagation of bending waves set up in the panel. Panel motion is greater near this frequency; therefore, noise transmission loss is less.

3. The sound coincidence occurs at a lower frequency in the composite panels resulting in a generally lower transmission loss in the critical speech interference range when compared to the aluminum panel.

The overall conclusion reached is that the lighter graphite-composite structure passes slightly more noise to the inside in the critical frequency range. To overcome this, a more effective internal treatment will be required to achieve a given noise criteria level.

3.2 WEIGHT ANALYSIS

Weight analysis and summaries are presented for the selected 0.98M ATT configuration. Standard AN group weight coding has been utilized for component and subsystem weights, as specified in MIL-STD-254 (ASG), given in all tables and discussions in the following subsections.

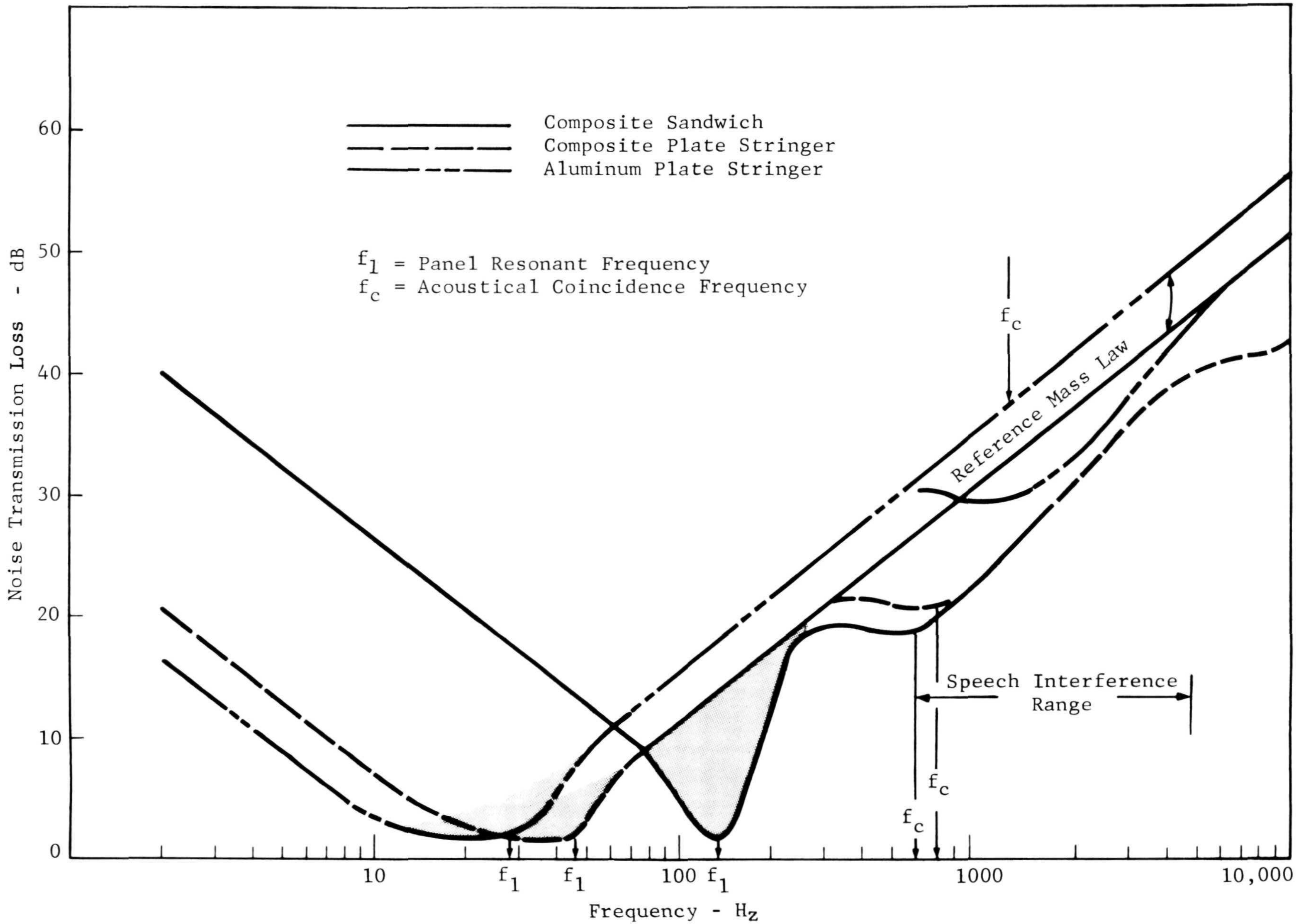


Figure 3.1-30 **Acoustic** Comparisons

3.2.1 Analysis Methods and Summaries

The weight summary for the M = .98 ATT selected configuration as reported in Reference 2-1 is shown in Table 3.2-1. These weights were obtained by interpolation of weight data generated for families of configurations. The process for obtaining the ATT family of configuration structural weights was to develop weights for the conventional aircraft materials (light alloy), to adjust these weights for composite (graphite) construction, and to re-adjust the weights for the effects of an active control system.

The basic aluminum (light-alloy) structural weights were calculated by a statistical-analytical procedure developed and continuously modified by General Dynamics through research studies. Table 3.2-2 shows by component, the basic parameters and the approximate number of equations used in the statistical-analytical procedure for a total airframe light-alloy structural weight buildup. The procedure in most part makes use of preliminary design aerodynamics, geometric parameters, and basic stress analyses. Because of the geometric extremes represented in some components of the advanced aircraft configurations, stress analyses were used extensively to reinforce and refine the statistical-analytical results. In some instances, flutter and aeroelastic analyses were performed to define the incremental wing weight necessary to meet stiffness requirements.

The procedure defined above allows component structural weights to be calculated for light-alloy (aluminum) only. To obtain structural weights for these components when designed with composite materials, a ratio of composite weight to light-alloy weight was applied to the basic light-alloy component weight. These ratios were obtained from stress/weight analyses performed on both composites and light-alloys for the various structural components, some on advanced aircraft configurations and some on other comparable structural studies (B-1 and F-111). The ratios applied to the .98M ATT selected configuration light-alloy weights to obtain composite structural weights are shown in Table 3.2-3.

The application of an active control system to the light-alloy- or composite-constructed airframe has been assumed to influence the structural weights of only the wings and fuselages. The structural weights of these components were determined by applying to the appropriate basic component weight (i.e., aluminum or composite), a ratio of the ACS

Table 3.2-1

ADVANCED TECHNOLOGY TRANSPORT
WEIGHT SUMMARY: MACH .98
COMPOSITES

Components	Pounds	Kilograms
Structure	(75,222)	(34,114)
Wing	26,300	11,927
Fuselage	28,075	12,732
Horizontal Tail	2,650	1,202
Vertical Tail	2,500	1,134
Landing Gear	11,227	5,092
Nacelles	4,470	2,027
Propulsion System	(19,189)	(8,702)
Engines	17,064	7,738
Water Injection System	265	120
Fuel System	1,525	692
Engine Controls	195	88
Starting Systems	140	64
Systems and Equipment	(42,353)	(19,208)
Surface Controls	4,210	1,910
Landing Gear Controls	1,318	598
Instruments	1,740	789
Hydraulics & Pneumatics	1,960	889
Electrical	3,217	1,459
Avionics	1,796	815
Furnishings	23,169	10,507
Air Conditioning	3,990	1,810
Auxiliary Gear	45	20
Auxiliary Power Unit	908	411
Weight Empty	136,764	62,024
Useful Load	(7,365)	(3,340)
Crew	1,430	649
Unusable Fuel	354	161
Engine Oil	120	54
Passenger Service	5,460	2,476
Operating Weight	144,128	65,364
Payload	40,000	18,140
Zero Fuel Weight	184,128	83,504
Fuel	88,752	40,251
Water	960	435
GROSS WEIGHT	273,840	124,190

WING W_{DES}, b', S_W t/c, λ, Λ, T, F_{TU} 15 Equations	+	**FUSELAGE** S_{WETT}, L_t, D, H, T, F_{TU}, Cutouts, Engines 29 Equations
+ **HORIZ. TAIL** F_H, b', S_{HT}, t/c, λ, Λ, T, F_{TU} 6 Equations	+	**VERT. TAIL** F_V, b', S_{VT} t/c, λ, Λ, T, F_{TU} 5 Equations
+ **LANDING GEAR** Percent of Gross Weight	+	**EXTERNAL NACELLE** S_N, S_{PYL}, W_{NC}, B_N, D_N, T, F_{TU} DC-10 Correlation + Engine Scaling

= TOTAL AIRFRAME

Table 3.2-2 Basic Parameters for Statistical-Analytical Weight Procedure

Table 3.2-3

STRUCTURAL WEIGHT RATIOS FOR
COMPOSITE AND ACTIVE CONTROL
SYSTEM EVALUATIONS

Component	Ratio		
	Composite Wt / Light-Alloy Wt	Light-Alloy ACS Wt / Light-Alloy Wt	Composite ACS Wt / Composite Wt
Wing	.72	.87	.91
Fuselage	.80	.98	.99
Horiz.Tail	.75	-	-
Vert.Tail	.80	-	-
Land.Gear	-	-	-
Nac./Pyl.	.90	-	-

component weight to the basic material component weight. These ratios were derived from the component stress/weight analyses and the stiffness requirements as determined by DAEAC active control system studies. The ratios for these components and materials are also shown in Table 3.2-3.

To verify the component weights obtained for the selected configuration by interpolation (Reference 2-1) and to obtain a statistical-analytical weight distribution by elements for each structural component, a weight analysis of the selected configuration was performed using the analysis process described above. The results of the weight analysis were not significantly different from the weights obtained by interpolation; the structural component weights were therefore not changed due to the statistical-analytical analysis performed. The results of the analysis were used, however, in conjunction with the DC 10-10 structural weight distributions and stress-derived structural element weights to develop a weight distribution by elements for each structural component. Summaries of the component weight distributions by elements are shown in Tables 3.2-4 through 3.2-8.

The critical design loads and the design criteria established in Reference 2-1 were utilized in conjunction with preliminary design methods to size both primary and secondary structural elements. Typical elements sized at sections using elementary beam theory are the wing surfaces and spars; the wing and vertical tail control surfaces and support structure; the fuselage skins, floors, and floor support beams; and the empennage surface and spars. In areas of concentrated load introduction, e.g., the nose and the main landing gear support structure; the control surface support structure; the horizontal tail support structure; and the wing fuselage intersection, analyses were made to estimate the critical load magnitudes and to size the support structure for load introduction and load redistribution.

The general approach used to establish the structural element distributions shown in Tables 3.2-4 through 3.2-8 is described as follows:

1. Determine those components of the light-alloy statistical weight distribution not likely to be affected by using composite construction.

2. Factor the component weights of the light-alloy statistical weight distribution not identified in Step 1 to adjust for composite construction.

TABLE 3.2-4

WING LIGHT-ALLOY/COMPOSITE WEIGHT BREAKDOWN

Structural Elements	Light-Alloy Element Wt.'s lb	kg	Composite Element Wt.'s lb	kg	Composite Airplane Breakdown: Material	Type of Construction	Manufacturing Process
Spar-Front/Outbd - B_L 370 to Tip						Sandwich	Premade Channels
Channels	250.9	113.8	157.0	71.2	Graphite		Bonded Back to Back
Core	8.6	3.9	8.6	3.9	3.0 lb/ft^3 Aluminum		on Core
Adhesive	13.7	6.2	13.7	6.2	Adhesive		
Spar - Front/Inbd - B_L 370 to C_L of A/C						Sandwich	Same as Spar Above
Channels	1182.4	536.3	748.0	339.3	Graphite		
Core	31.2	14.2	31.2	14.2	3.0 lb/ft^3 Aluminum		
Adhesive	50.0	22.7	50.0	22.7	Adhesive		
Spar - Rear - C_L of A/C to Tip						Sandwich	Same as Spar Above
Channels	2629.9	1192.9	1705.0	773.4	Graphite		
Core	23.3	10.6	23.3	10.5	3.0 lb/ft^3 Aluminum		
Adhesive	37.2	16.9	37.2	16.9	Adhesive		
Ribs & Blkhds. from C_L of A/C						Stiffened Sheet Ribs	Solid Laminate with
1	178.3	80.9	114.9	52.1	Graphite Sheet Fiberglass Stiffeners		Premade Stiffeners
2	170.7	77.4	110.0	49.9	" " " "		Attached
Pressure Blkhd.	472.7	214.4	304.6	138.1	Graphite Skins Aluminum Core	Sandwich	Co-Cured
3	122.8	55.7	79.1	35.9	Graphite Sheet Fiberglass Stiffeners		
4	104.0	47.2	67.0	30.4	" " " "		
5	88.8	40.3	57.3	26.0	" " " "		
Pressure Blkhd.	223.4	101.3	144.0	65.3	Graphite Skins Aluminum Core	Sandwich	Co-Cured
6	59.6	27.0	38.4	17.4	Graphite Sheet Fiberglass Stiffeners		
7	50.7	23.0	32.7	14.8	" " " "		
8	37.2	16.9	24.0	10.9	" " " "		
9	30.0	13.6	19.3	8.7	" " " "		
10	21.7	9.8	14.0	6.3	" " " "		
11	19.7	8.9	12.7	5.8	" " " "		
12	18.9	8.6	12.2	5.5	" " " "		
13	18.2	8.3	11.7	5.3	" " " "		
14	17.7	8.0	11.4	5.2	" " " "		
15	16.6	7.5	10.7	4.8	" " " "		
16	16.5	7.5	10.6	4.8	" " " "		
17	15.4	7.0	9.9	4.5	" " " "		

TABLE 3.2-4
(Continued)

WING LIGHT-ALLOY/COMPOSITE WEIGHT BREAKDOWN

Structural Elements	Light-Alloy Element Wt.'s lb	kg	Composite Element Wt.'s lb	kg	Composite Airplane Breakdown Material	Type of Construction	Manufacturing Process
Ribs (Cont'd)							
18	15.1	6.8	9.7	4.4	Graphite Sheet Fiberglass Stiffeners		
19	14.9	6.8	9.6	4.3	" " " "		
20	13.8	6.3	8.9	4.0	" " " "		
21	13.3	6.0	8.6	3.9	" " " "		
Pressure Blkhd.	16.5	7.5	10.6	4.8	Graphite Skins Aluminum Core	Sandwich	Co-Cured
22	12.3	5.6	7.9	3.6	Graphite Sheet Fiberglass Stiffeners		
23	12.0	5.4	7.7	3.5	" " " "		
24	11.5	5.2	7.4	3.4	" " " "		
25	10.7	4.9	6.9	3.1	" " " "		
26	10.6	4.8	6.8	3.1	" " " "		
Pressure Blkhd.	14.0	6.4	9.0	4.1	" " " "		
27	9.8	4.4	6.3	2.9	" " " "		
28	9.6	4.4	6.2	2.8	" " " "		
29	9.2	4.2	5.9	2.7	" " "		
Tip Closing Blkhd.	64.0	29.0	64.0	29.0	Steel	Machined Forging	
Skin-Upper - Tip to Tip						Sandwich	Pre-made Outer Surface Skin-Core Bonded with Wet Inner Skin
Outer Covering	2463.8	1117.6	1475.0	669.1	Graphite		
Core	595.0	269.9	595.0	269.9	3.0 lb/ft^3 Aluminum		
Adhesive	230.0	104.3	230.0	104.3	Adhesive		
Inner Covering	2463.8	1117.6	1475.0	669.1	Graphite		
Skin-Lower - Tip to Tip						Sandwich	Same as Skin Above
Outer Covering	2669.6	1210.9	1610.0	730.3	Graphite		
Core	595.0	269.9	595.0	269.9	3.0 lb/ft^3 Aluminum		
Adhesive	230.0	104.3	230.0	104.3	Adhesive		
Inner Covering	2669.6	1210.9	1610.0	730.3	Graphite		
Wing Box Attach Fasteners	1070.8	485.7	639.0	289.9	Misc.		
Trail. Edge Fittings	340.0	154.2	340.0	154.2	Titanium		Machined
Pylon Attach Fittings	326.0	147.9	326.0	147.9	Titanium		Machined
Access Doors & Fasteners	320.0	145.2	210.0	95.3	Graphite/Core	Sandwich	Co-Cured
TOTAL STRUCTURAL BOX WEIGHT	20121.0	9126.9	13369.0	6064.2			

TABLE 3.2-4
(Continued)

WING LIGHT-ALLOY/COMPOSITE WEIGHT BREAKDOWN

Structural Elements	Light Alloy Element Weights lb.	kg.	Composite Element Weights lb.	kg.	Composite Airplane Construction: Material	Type of Construction	Manufacturing Process
Aileron							
Covering & Stiffeners	436.0	197.8	371.0	168.3	Graphite - Al. Core	Sandwich	Premade Outer Skin & Other
Spar	47.0	21.3	40.0	18.1	Graphite	Solid Laminate	Details Bonded to Wet
T.E. Closure	10.0	4.5	10.0	4.5	Aluminum	Extrusion	Innerskin & Core
Carriage & Support	109.1	49.5	109.1	49.5	Steel	Machined	
Misc.	1.9	0.9	1.9	0.9			
Flaperon							
Covering	517.8	234.9	440.0	199.5	Graphite - Al. Core	Sandwich	Same As Aileron Above
Spar	63.6	28.8	54.0	24.5	Graphite	Solid Laminate	
Rib	70.6	32.0	60.0	27.2	Graphite	Solid Laminate	
T.E. Closure	14.0	6.3	14.0	6.3	Aluminum	Extrusion	
Carriage & Support	147.0	66.7	147.0	66.7	Steel	Machined	
Misc.	2.0	0.9	2.0	0.9			
Flaps					(Al. Wt. Total for 6 Flaps - Composite Wt. Total for 8 Flaps)		
Covering	1225.0	555.7	925.6	419.9	Graphite - Al. Core	Sandwich	Same as Aileron Above
Spar	774.2	351.2	585.0	265.4	Graphite	Solid Laminate	
Rib	839.1	380.6	634.0	287.6	Graphite	Solid Laminate	
T.E. Closure	45.6	20.7	45.6	20.7	Aluminum	Extrusion	
Actuator Fairing	354.0	160.6	354.0	160.6	Fiberglass	Solid Laminate	
Actuator & Mechanism	890.6	404.0	890.6	404.0	Misc.	Misc.	
Track & Support	1645.2	746.3	1645.2	746.3	Steel	Machined	
Misc.	13.3	6.0	13.3	6.0	Misc.		
Vanes							
Covering	746.3	338.5	634.0	287.6	Graphite - Al . Core	Sandwich	Same as Aileron Above
Spar	110.7	50.2	94.0	42.6	Graphite	Solid Laminate	
T.E. Closure	46.0	20.8	46.0	20.8	Aluminum	Extrusion	
Actuator Fairing	48.0	21.8	48.0	21.8	Fiberglass	Solid Laminate	
Actuator & Mechanism	42.0	19.0	42.0	19.1	Misc.	Misc.	
Track & Support	80.0	36.3	80.0	36.3	Steel	Machined	

TABLE 3.2-4
(Continued)

WING LIGHT-ALLOY/COMPOSITE WEIGHT BREAKDOWN

Structural Elements	Light-Alloy Element Weight lb	Light-Alloy Element Weight kg	Composite Element Weight lb	Composite Element Weight kg	Composite Airplane Construction: Material	Composite Airplane Construction: Type of Construction	Composite Airplane Construction: Manufacturing Process
Doors - Deflector & Access							
Covering	99.0	44.9	65.7	29.8	Graphite - Al. Core	Sandwich	Co-Cure Frame Core & Skins
Frame & Hinge	120.0	54.4	120.0	54.4	Aluminum	Machined	
Hinge Fairing	8.0	3.6	8.0	3.6	Fiberglass	Solid Laminate	
Hinge Support	50.0	22.7	50.0	22.7	Aluminum	Machined	
Slats							
Covering & Stiffeners	421.5	191.2	336.8	166.4	Graphite - Fiberglass	Solid Laminate	
Air Director Node	113.5	51.5	98.8	44.8	Graphite - Al. Core	Sandwich	Co-Cure
Ribs	218.6	99.2	190.2	86.3	Graphite	Solid Laminate	
Actuator & Mechanism	389.3	176.6	389.3	176.6	Misc.	Misc.	
Actuator Support	632.7	287.0	550.5	249.7	Graphite	Solid Laminate	
Misc	22.4	10.2	22.4	10.2	Misc.	Misc.	
Spoilers					(Al. Wt. Total for 10 Spoilers - Composite Wt. Total for 14 Spoilers)		
Covering	636.0	288.5	509.0	230.9	Graphite - Al. Core	Full Depth Sandwich	Same as Aileron Above
Actuator & Mechanism	208.0	94.3	208.0	94.3	Misc.	Misc.	
TOTAL WING CONTROL SURFACE WEIGHT	11198.0	5079.4	9865.0	4474.8			

TABLE 3.2-4
(Continued)

WING LIGHT-ALLOY/COMPOSITE WEIGHT BREAKDOWN

Structural Elements	Light-Alloy Element Weights lb.	Light-Alloy Element Weights kg.	Composite Element Weights lb.	Composite Element Weights kg.	Composite Airplane Breakdown: Material	Composite Airplane Breakdown: Type of Construction	Composite Airplane Breakdown: Manufacturing Process
Fixed Leading Edge	1277.0	579.2	1149.0	521.2	Graphite - Al. Core	Core Stiffened Sheet	Co-Cure
Fixed Trailing Edge	752.0	341.1	639.0	289.8	Graphite - Al. Core	Core Stiffened Sheet	Co-Cure
MLG. Doors	317.0	143.8	317.0	143.8	Graphite - Al. Core	Full Depth Sandwich	Premade Outer Skin Bonded to Core & Wet Inner Skin
Tip	125.0	56.7	125.0	56.7	Graphite - Al. Core	Full Depth Sandwich	Premade Outer Skins Bonded to Core
Wing Fuselage Fillet							
Covering	434.2	197.0	369.0	167.4	Graphite - Al. Core	Core Stiffened Sheet	Co-Cure
Ribs	258.8	117.4	220.0	99.8	Graphite - F/G	F/G Stiffened Sheet	Co-Cure
Auxiliary Spar	166.0	75.3	108.0	49.0	Graphite	Solid Laminate	
Misc.	139.0	63.1	139.0	63.0			
TOTAL WING SECONDARY STRUCTURAL WEIGHT	3469.0	1573.6	3066.0	1390.7			
TOTAL WING WEIGHT	34788.0	15779.8	26300.0	11929.7			

TABLE 3.2-5

FUSELAGE LIGHT-ALLOY/COMPOSITE WEIGHT BREAKDOWN

Structural Elements	Light-Alloy Element Weights lb.	Light-Alloy Element Weights kg.	Composite Element Weights lb.	Composite Element Weights kg.	Composite Airplane Breakdown: Material	Composite Airplane Breakdown: Type of Construction	Composite Airplane Breakdown: Manufacturing Process
Fuselage Skins							
- Upper Right Nose Segment	271.2	123.0	190.3	86.3	Graphite	Sandwich	Co-Cured Sandwich
- Upper Left Nose Segment	271.2	123.0	190.3	86.3	"	"	" "
- Lower Nose Segment	271.3	123.0	190.4	86.3	"	"	" "
- Upper Right Body Segment	3478.0	1577.6	2434.6	1104.3	"	"	" "
- Upper Left Body Segment	3478.0	1577.6	2434.6	1104.3	"	"	" "
- Lower Body Segment	3478.0	1577.6	2434.6	1104.3	"	"	" "
- Right Tail Section	748.6	339.6	524.1	273.7	"	"	" "
- Left Tail Section	748.7	339.6	524.1	273.7	"	"	" "
Skin Splices	610.0	276.7	610.0	276.9	Graphite	Laminate	Vacuum Bag
Longerons	898.0	407.3	628.0	284.9	Graphite	Laminate	Co-Cured
Window Framing	1444.0	655.0	1008.0	457.2	Graphite	Laminate	Co-Cured
Frames (98)	3098.0	1405.3	2163.0	981.1	Graphite/Fiberglass	Laminate	Vacuum Bag
Bulkhead							
- Aft of Nose Cone	30.0	13.6	30.0	13.6			
- Fwd. of Nose Gear Bay	85.0	38.6	85.0	38.6	Graphite	Sandwich	Co-Cured
- Aft of Nose Gear Bay	130.0	59.0	130.0	59.0	"	"	"
- Fwd. of Wing Box	135.0	61.2	135.0	61.2	Graphite/Boron	Laminate	Vacuum Bag
- Aft of Wing Box	140.0	63.5	140.0	63.5	" "	"	"
- Aft of Main Landing Gear Bay	130.0	59.0	130.0	59.0	" "	Sandwich	Co-Cured
- Aft Pressure	315.0	142.9	315.0	142.9	Graphite	Laminate	Vacuum Bag
- Fwd. of H.T. Box	75.0	34.0	75.0	34.0	"	"	"
- Aft of H.T. Box	75.0	34.0	75.0	34.0	"	"	"
- V.T. Front Spar Support	70.0	31.8	70.0	31.8	"	Sandwich	Co-Cured
- V.T. Front Intermediate Spar Support	140.0	63.5	140.0	63.5	"	"	"
- V.T. Rear Intermediate Spar Support	225.0	102.1	225.0	102.1	"	"	"
- V.T. Rear Spar Support	170.0	77.1	170.0	77.1	"	"	"
- Tail Cone Support	30.0	13.6	30.0	13.6	"	Laminate	Vacuum Bag
Misc.	397.0	180.1	397.0	180.1			
TOTAL PRIMARY STRUCTURAL WEIGHT	20942.0	9499.3	15479.0	7021.3			

TABLE 3.2-5
(Continued)

FUSELAGE LIGHT-ALLOY/COMPOSITE WEIGHT BREAKDOWN

Structural Elements	Light-Alloy Element Weights lb.	Light-Alloy Element Weights kg.	Composite Element Weights lb.	Composite Element Weights kg.	Composite Airplane Breakdown: Material	Composite Airplane Breakdown: Type of Construction	Composite Airplane Breakdown: Manufacturing Process
Floors - Cockpit	73.0	33.1	59.0	26.8	Graphite	Sandwich	Co-Cure
- Cabin	2936.0	1331.8	2378.0	1078.6	"	"	"
- Cargo Container	380.0	172.4	308.0	139.7	"	"	"
- Bulk Cargo	188.0	85.3	152.0	68.9	"	"	"
Floor Supports - Cockpit	70.1	31.8	57.0	25.9	Graphite	Laminate	Vacuum Bag
- Cabin	1062.0	481.7	860.0	390.1	"	"	" "
- Cargo Container	81.0	36.7	66.0	29.9	"	"	" "
- Bulk Cargo	37.0	16.8	30.0	13.6	"	"	" "
Floor Tracks - Cabin Seats	88.0	39.9	71.0	32.2	Aluminum	Machined	Conventional
- Cargo Rollers	305.0	138.3	243.0	110.2	Steel	"	"
Doors - Passenger (6)	418.0	189.6	349.0	158.3	Graphite	Sandwich	Co-Cured
- Fwd Upper Cargo	55.0	24.9	46.0	20.9	"	"	"
- Fwd. Lower Cargo	135.0	61.2	113.0	51.3	"	"	"
- Aft Cargo	123.0	55.8	103.0	46.7	"	"	"
- Landing - MLG	449.0	203.7	375.0	170.1	"	"	"
- Landing - NLG	161.0	73.0	134.0	60.8	"	"	"
- Air Conditioning	58.0	26.3	48.0	21.8	"	"	"
- Access - Flight Deck	12.0	5.4	10.0	4.5	"	"	"
- Access - Avionics	19.0	8.6	16.0	7.3	"	"	"
- Access - Tail Section	22.0	10.0	18.0	8.2	"	"	"
- Access - NLG Well Sidewalls	98.0	44.5	82.0	37.2	"	"	"
Door Opening Mech - Passenger	1547.0	701.7	1547.0	701.7	Aluminum/Steel	Machined	Conventional
- Fwd Upper Cargo	10.0	4.5	10.0	4.5	" "	"	"
- Fwd Lower Cargo	359.0	162.8	359.0	162.8	" "	"	"
- Aft Cargo	223.0	101.2	223.0	101.2	" "	"	"
- Landing - MLG	136.0	61.7	136.0	61.7	" "	"	"
- Landing - NLG	25.0	11.3	25.0	11.3	" "	"	"
- Air Conditioning	17.0	7.7	17.0	7.7	" "	"	"

Table 3.2-5 (Continued)

FUSELAGE LIGHT-ALLOY/COMPOSITE WEIGHT BREAKDOWN

STRUCTURAL ELEMENTS	LIGHT-ALLOY ELEMENT WEIGHTS		COMPOSITE ELEMENT WEIGHTS		COMPOSITE AIRPLANE BREAKDOWN		
	LB.	KG.	LB.	KG.	MATERIAL	TYPE OF CONSTRUCTION	MANUFACTURING PROCESS
Door Frames - Passenger	744.0	337.5	621.0	281.7	Graphite	Laminate	Co-Cured
- Fwd. Upper Cargo	110.0	49.9	92.0	41.7	"	"	"
- Fwd. Lower Cargo	162.0	73.5	135.0	61.2	"	"	"
- Aft Cargo	155.0	70.3	129.0	58.5	"	"	"
- Landing - MLG	210.0	95.3	175.0	79.4	"	"	"
- Landing - NLG	185.0	83.9	154.0	69.9	"	"	"
Passenger Window Glass	1560.0	707.6	1560.0	707.6			
Windshield & Canopy	475.0	215.5	475.0	215.5			
Nose Structure & Radome	100.0	45.4	100.0	45.4	Fiberglass	Laminate	Vacuum Bag
Tail Cone - Structure	543.0	246.3	543.0	246.3	Graphite	Laminate	Vacuum Bag
- Operating Mechanism	43.0	19.5	43.0	19.5	Aluminum/Steel		Conventional
Primer & Sealent	437.0	198.2	437.0	198.2	Miscellaneous		
Exterior Finish	78.0	35.4	78.0	35.4	"		
Drains	41.0	18.6	41.0	18.6	"		
Walkways, Steps, & Grips	28.0	12.7	28.0	12.7	"		
Misc.	150.0	68.1	150.0	68.0	"		
Total Secondary Structure Weight	14108.0	6399.4	12596.0	5713.5			
Total Fuselage Structure Weight	35050.0	15898.7	28075.0	12734.8			

Table 3.2-6

VERTICAL TAIL LIGHT-ALLOY/COMPOSITE WEIGHT BREAKDOWN

Structural Element	Light-Alloy Element Weight		Composite Element Weight		Composite Airplane Breakdown		
	lb	kg	lb	kg	Material	Type of Construction	Manufacturing Process
Spar Caps							
Front	292.0	132.5	204.0	92.5	Graphite	Solid Laminate	Vacuum Bag
2nd	218.0	98.9	152.5	69.2	Graphite	Solid Laminate	Vacuum Bag
3rd	213.0	96.6	149.6	67.9	Graphite	Solid Laminate	Vacuum Bag
Rear	242.0	109.8	169.9	77.1	Graphite	Solid Laminate	Vacuum Bag
Skin	647.5	293.7	452.0	205.0	Graphite Skin Al. Core	Sandwich	Co-Cured
Spar Webs							
Front	49.6	22.5	49.6	22.5	Graphite Skin Al. Core	Laminate/Sandwich	Co-Cured
2nd	42.0	19.1	42.0	19.1	" "	" "	"
3rd	42.6	19.3	42.6	19.3	" "	" "	"
Rear	42.8	19.4	42.8	19.4	" "	" "	"
Ribs	276.0	125.2	207.0	93.9	" "	Sandwich	"
Leading Edge							
Skin	73.8	33.5	62.5	28.4	Graphite Skin Al. Core	Sandwich	Co-Cured
Ribs	32.4	14.7	27.5	12.5	" "	"	"
Joints & Doors	62.8	28.5	54.0	24.5	Graphite	Laminate	Pre-Cured
Trailing Edge							
Skin	14.2	6.4	12.0	5.4	Graphite Skin Al. Core	Sandwich	Co-Cured
Ribs	5.3	2.4	5.3	2.4	Al-Aly	Machined	
Joints	10.7	4.9	10.7	4.9			
Rudder	530.0	240.4	450.0	204.1	Graphite Skins Al. Core	Sandwich	Co-Cured
Vert-Fus Attach	93.0	42.2	65.0	29.5	Glass-Prepreg	Wet Layup	
Misc.	237.3	107.6	301.0	136.5	Finish-Fairings Rudder, Etc.		
TOTAL	3125.0	1417.6	2500.0	1134.1			

NOTE: Spar weights reflect material above the vertical tail-fuselage intersection although spar design is integral with fuselage bulkhead.

TABLE 3.2-7

HORIZONTAL TAIL LIGHT-ALLOY/COMPOSITE WEIGHT BREAKDOWN

Structural Element	Light-Alloy Element Weights lb.	kg.	Composite Element Weights lb.	kg.	Composite Airplane Breakdown: Material	Type of Construction	Manufacturing Process
Upper Cover						Sandwich	Premade Outer
Skins	617.9	280.3	442.5	200.7	Graphite		Skin, Co-
Core	221.3	100.4	158.5	71.9	3# Aluminum		Curing Inner
Adhesive	76.5	34.7	76.5	34.7	Adhesive		Skin and Core
Lower Cover						Sandwich	Same as Above
Skins	617.9	280.3	442.5	200.7	Graphite		
Core	221.3	100.4	158.5	71.9	3# Alum.		
Adhesive	76.5	34.7	76.5	34.7	Adhesive		
Front Spar						Sheet-Stiffened	Rivet/Bond
Caps	72.9	33.1	52.0	23.6	Graphite		Assembly
Web	21.3	9.7	15.2	6.9	Graphite		
Stiffeners	9.6	4.4	9.6	4.4	Fiberglass		
Spar 2						Sandwich	Premade
Caps	86.9	39.4	62.9	28.5	Graphite		Channel, Co-
Webs	55.8	25.3	40.4	18.3	Graphite		Cure Core &
Core	27.4	12.4	19.8	9.0	3# Alum.		Opposite
Adhesive	10.6	4.8	10.6	4.8	Adhesive		Channel
Spar 3						Sandwich	Same as
Caps	70.9	32.2	51.3	23.3	Graphite		Spar 2
Webs	45.3	20.5	32.8	14.9	Graphite		
Core	22.3	10.1	16.1	7.3	3# Alum.		
Adhesive	8.6	3.9	8.6	3.9	Adhesive		
Rear Spar						Sheet-Stiffener	Rivet/Bond
Caps	62.0	28.1	45.9	20.8	Graphite		Assembly
Web	19.6	8.9	14.5	6.6	Graphite		
Stiffeners	12.2	5.5	9.0	4.1	Fiberglass		
Leading Edge						Sheet-Stiffener	
Skin	84.7	38.4	65.5	29.7	Graphite		
Stiffeners	21.0	9.5	16.2	7.3	Fiberglass		
Adhesive	1.3	0.6	1.3	0.6	Adhesive		

TABLE 3.2-7
(Continued)

HORIZONTAL TAIL LIGHT-ALLOY/COMPOSITE WEIGHT BREAKDOWN

Item	Light-Alloy Element Weight		Composite Element Weight		Composite Airplane Breakdown		
	lb	kg	lb	kg	Material	Type of Construction	Manufacturing Process
Trailing Edge							
Upper Cover	71.3	32.3	54.9	24.9	Graphite w/Fiberglass Stiffeners	Sheet Stiffener	Rivet/Bond
Lower Cover	71.3	32.3	54.9	24.9	" " "	" "	Assembly
Ribs	57.4	26.0	44.2	20.0	Graphite w/Graphite Stiffeners	" "	" "
Wing Tip							Premade Upr.
Skins	39.4	17.9	27.9	12.7	Graphite	Full Depth Core	& Lwr. Skins
Core	76.9	34.9	54.4	24.7	3# Aluminum		Bonded Core
Adhesive	14.2	6.4	14.2	6.4			Installation
Inbd. Box Rib	80.6	36.6	59.6	27.0	Graphite	Sheet Stiffener	Rivet/Bond Assembly
Inbd. Closure Rib	64.7	29.3	47.9	21.7	"	" "	" "
Rib 1	46.4	21.1	34.3	15.6	"	" "	" "
Rib 2	39.6	18.0	29.3	13.3	Graphite w/Aluminum Core	Sandwich	Same as Spar 2
Rib 3	33.2	15.1	24.6	11.2	Graphite	Sheet Stiffener	Rivet/Bond Assembly
Rib 4	27.6	12.5	20.4	9.3	"	" "	" "
Rib 5	22.4	10.2	16.6	7.5	"	" "	" "
Rib 6	17.8	8.0	13.2	6.0	"	" "	" "
Tip Closure Rib	13.4	6.1	9.9	4.5	"	Sheet Stiffener	" "
Hinges & Actuator	58.0	26.3	59.0	26.3			
Fasteners	297.0	134.7	220.0	99.8	Steel		
Miscellaneous	39.0	17.7	39.0	17.7			
Total Weight	3534.0	1603.0	2650.0	1202.1			

Table 3.2-8

NACELLE LIGHT-ALLOY/COMPOSITE WEIGHT BREAKDOWN

Structural Element	Light-Alloy Element Weight lb	Light-Alloy Element Weight kg	Composite Element Weight lb	Composite Element Weight kg	Composite Airplane Breakdown: Material	Composite Airplane Breakdown: Construction Concept	Composite Airplane Breakdown: Manufacturing Process
Engine Mounts	127.0	57.6	127.0	57.6	Steel	Machined	
Pylon & Struts*	1500.0	680.4	1500.0	680.4	Steel	Welded	
Cowling	3340.0	1515.0	2843.0	1289.6	Graphite Skins, Nomex Core	Honeycomb Sandwich	Co-Cured
TOTAL NACELLE WEIGHT	4967.0	2253.0	4470.0	2027.6			

*The pylon and struts weight is for the wing nacelles only. The vertical tail pylon and struts weight is included in the vertical tail misc. weights.

3. Distribute the component weights identified in Steps 1 and 2 to the structural elements of each component utilizing preliminary element sizing results, element geometry, and the element weight distribution for the DC-10-10.

A summary of the component weights by material types is presented in Table 3.2-9. The data in Table 3.2-9 show that 68 percent of the airframe structural weight considered in this study is composite material.

3.2.2 Concept Studies

Two different structural concepts in combination with aluminum and composite materials were analyzed to permit structural concept and material selection weight comparisons. The concepts analyzed included (1) graphite sandwich, (2) graphite sheet-stringer, (3) aluminum sandwich, and (4) aluminum-skins with graphite stiffeners.

Two sections inboard and two sections outboard of the wing expanded contour break-line of the two spar wing box structure of the selected configuration were analyzed. It is assumed that the trends determined from the analysis of these sections will also be indicative of trends in other areas of application such as the fuselage and empennage surfaces.

In the analysis of the different concepts the working stress levels for aluminum construction established in Reference 2-1 for a dependable, long-life aircraft were utilized. For the composite skin construction, three inch wide buffer strips composed of plies oriented plus and minus forty-five degrees (0.79 Rad.) with the spar caps are used for a long-life design in the areas where the design requires mechanical fasteners. It is assumed in the analysis of the two composite skin concepts that the buffer strips do not contribute to the reaction of bending loads.

The results of the analysis are shown in Figure 3.2-1. The aluminum sandwich covers have to be eliminated from consideration because the required skin thicknesses exceeded practical manufacturing limitations. Based on the average of the box section weights plotted in Figure 3.2-1 the composite sheet-stiffener construction is fourteen percent lighter than the composite sandwich construction and the aluminum-skin composite-stiffener cover construction thirty-eight percent heavier.

Table 3.2-9

COMPONENT MATERIAL WEIGHT SUMMARY

Component lb (kg)		Material lb (kg)		
		Graphite and Fiberglass	Aluminum Core and Adhesive	Steel, Titanium, Aluminum and Fasteners
Wing	26,300 (11,920)	17,800 (8,070)	3,100 (1,408)	5,400 (2,450)
Fuselage	28,075 (12,710)	19,787 (8,950)	2,448 (1,112)	5,840 (2,642)
Tails	5,150 (2,339)	3,640 (1,650)	870 (394)	640 (290)
Nacelles	4,470 (2,025)	2,270 (1,029)	573 (260)	1,627 (737)
Total	63,995 (28,994)	43,497 (19,699)	6,991 (3,174)	13,507 (6,119)
	100%	68%	11%	21%

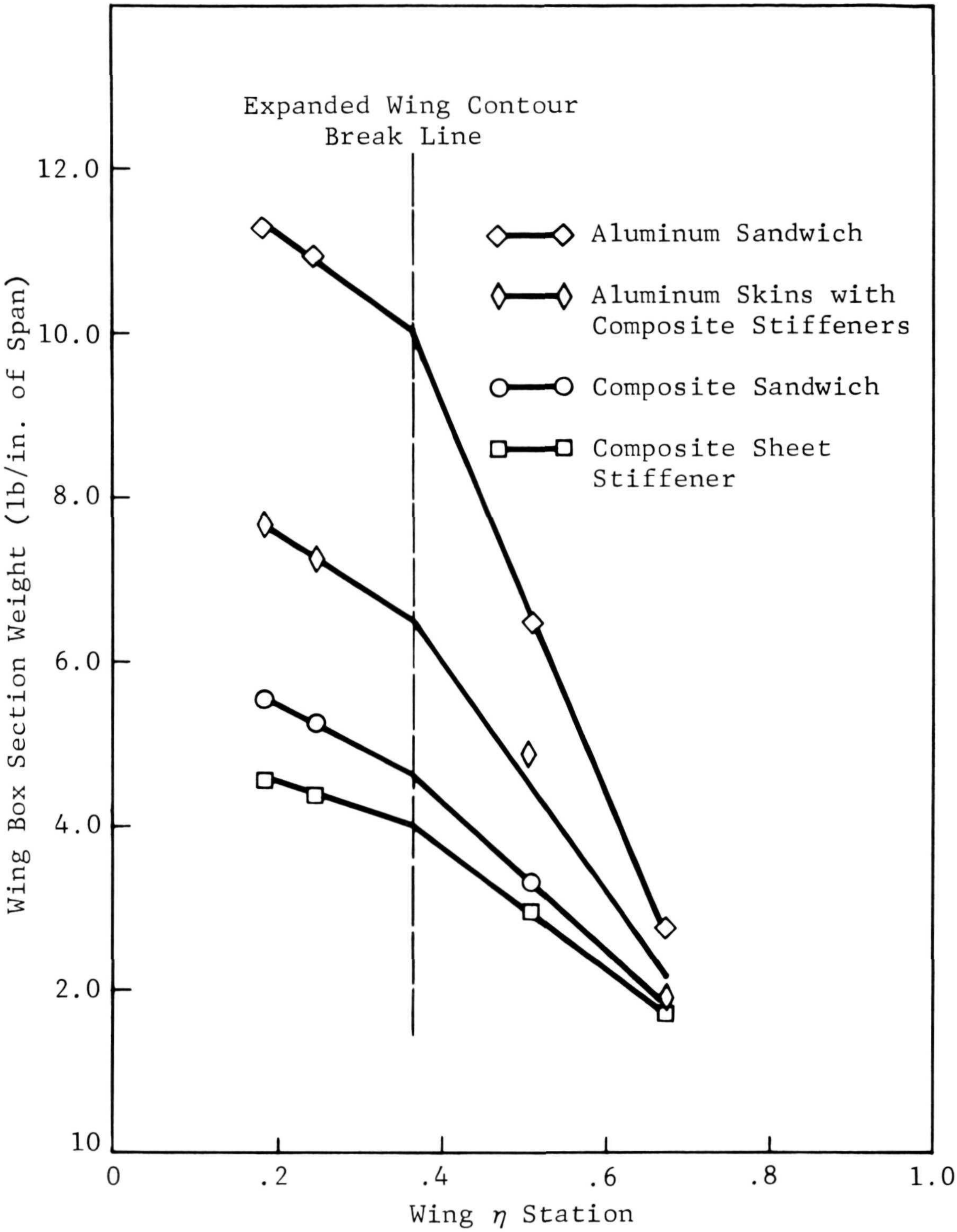


Figure 3.2-1 Cover Weight Comparisons

The sheet stiffened minimum weight designs are two inches (0.051 m) in depth, have two inch (0.051 m) wide caps, and have an average spacing of four and one-half inches (0.114 m) for the sections analyzed. The minimum weight composite sandwich design is a two inch (0.051 m) thick, three-pound per cubic foot (48.1 kg per cubic m) expanded aluminum core construction. The composite skins analyzed are assumed to be sixty percent of zero degree graphite plies and forty percent plus and minus forty-five degree (0.79 Rad.) graphite plies. The cap material of the composite stiffeners is assumed to be eighty percent zero degree graphite plies and twenty percent plus and minus forty-five degree (0.79 Rad.) graphite plies. The composite stiffener webs are assumed to be one-hundred percent plus and minus forty-five degree (0.79 Rad.) graphite plies.

The composite sheet construction is lighter than the composite sandwich construction because the effective bending material of the composite sheet stringer construction is located farther from the neutral axis than is the effective bending material of the composite sandwich for the sections analyzed. Also, the cap material of the sheet stringer has higher tension and compression stress allowables than the skin material in the direction of the primary bending loads since there is a higher percentage of zero degree fibers in the stiffener caps than in the composite skins. In addition, the sandwich construction has to pay the penalty of having buffer strips in both the inner and outer skins whereas the sheet stringer construction has to pay the penalty of only one buffer strip.

SECTION 4

MANUFACTURING APPROACH

The major features of the approach envisioned for the manufacture of the advanced composite transport airframe structure are discussed in this section.

4.1 BASIC ASSUMPTIONS

The assumed schedules of the three major manufacturing activities are shown in Figure 4.1-1 along with the corresponding schedule of aircraft produced. The delivery rate for the 250 unit production program was assumed to reach a maximum of six aircraft per month.

From a cost and reliability standpoint it is necessary to manufacture very large pieces of structure. To do this at reasonable cost mechanized lay-up capability is required.

Mechanized lay-up equipment requires the use of composite materials in preimpregnated continuous tape form. The tape system must be supplied with very little excess resin to minimize the outflow during the curing process. The prepreg materials should require no more than vacuum bag curing pressures and no more than 250^{o}F (394^{o}K) so that simple ovens may be used instead of autoclaves.

Other manufacturing requirements are co-curing of details (some with unsymmetrical sections) one-step processing of bonding tools from tooling masterforms, and tools and bagging materials with integral heating devices.

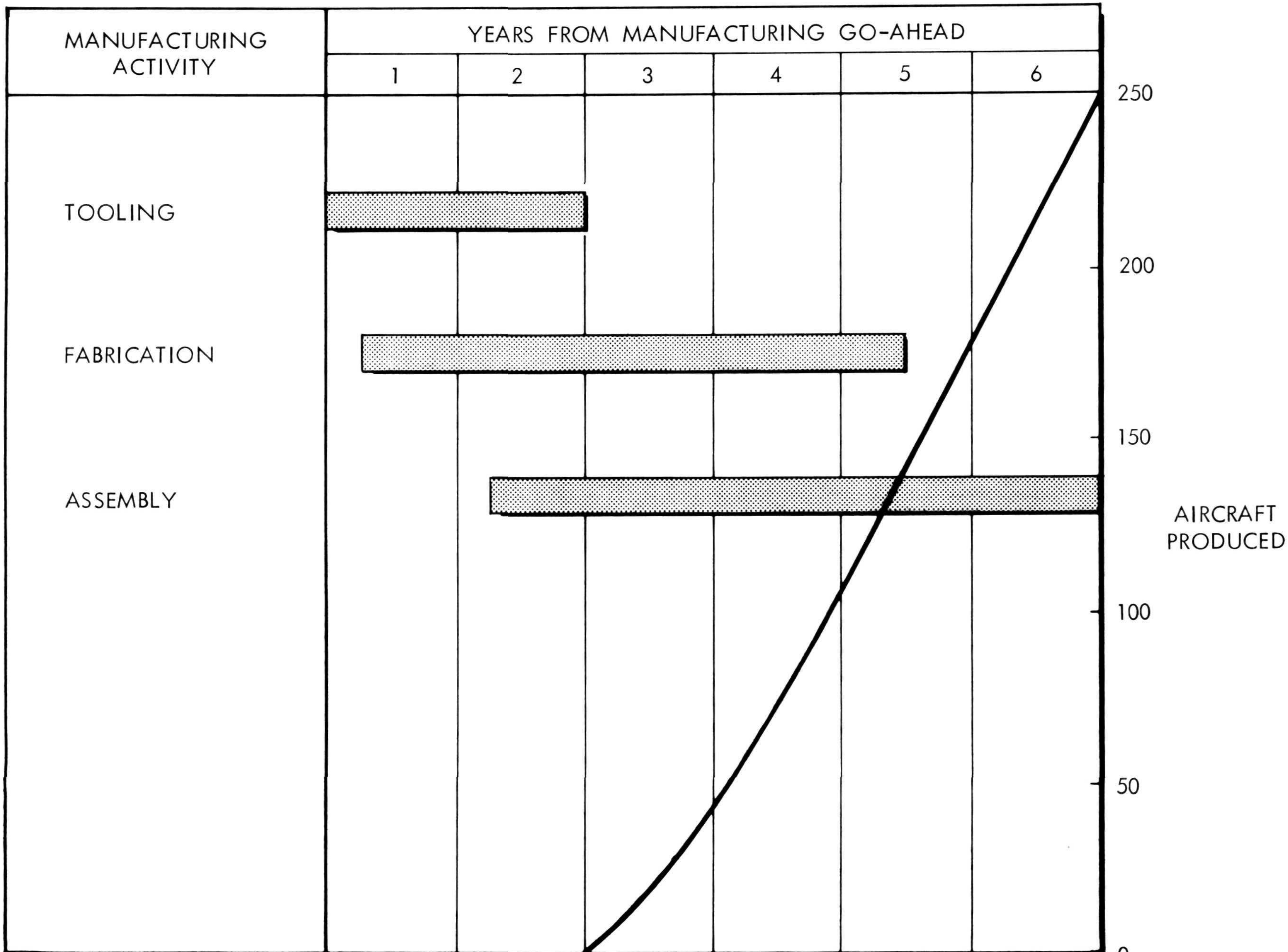


Figure 4.1-1 Manufacturing Schedule

4.2 TOOLING APPROACH

The tooling approach for the transport will employ extensive use of composite materials, mostly fiberglass, in the tool surfaces. The major tools used to fabricate the surface structure with the various required contours will themselves be fabricated directly from conventional plaster master forms in one step as depicted in Figure 4.2-1. The manufacturing approach and scheduled delivery rates will require several duplications of some of these tools and each will be fabricated as shown.

Many of the more ordinary sized parts with small contour changes will utilize aluminum or steel tooling. These parts will include channels and stiffeners which are precured before assembly.

Another type of tooling to be utilized will provide the source for the secondary bonding pressures and temperatures as shown in Figure 4.2-2. The pliable bladders will be easily handled, inflated to provide the bonding pressures, and contain integral heater elements to provide the bonding temperatures.

4.3 FABRICATION APPROACH

The manufacturing approach for the composite airframe emphasizes the fabrication and assembly of very large parts. Automation and mechanization are utilized in every phase. In addition, the mass production concept is emphasized. Parts move from area to area while undergoing transformations from detail fabricated parts to elements of assemblies. The

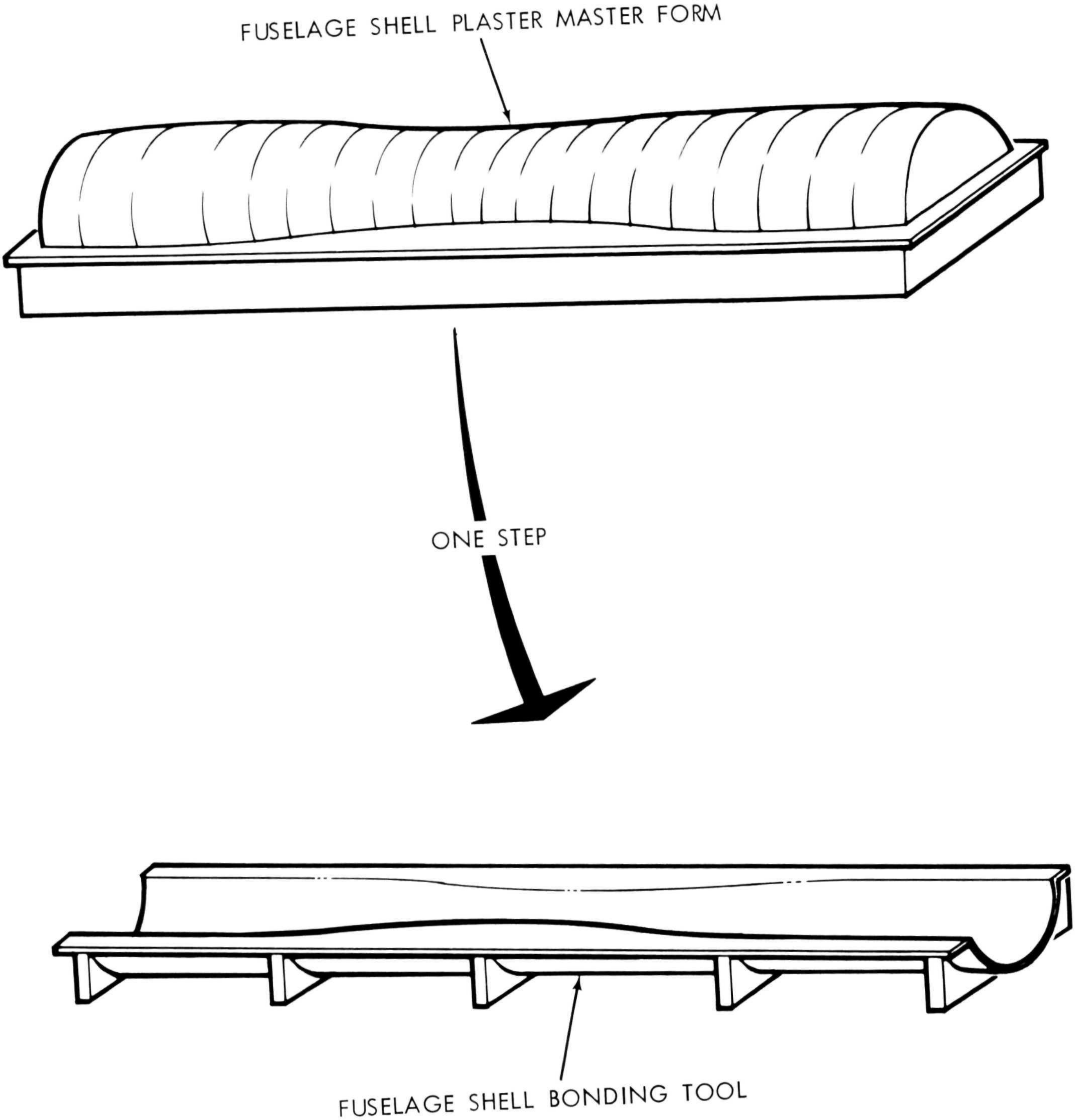
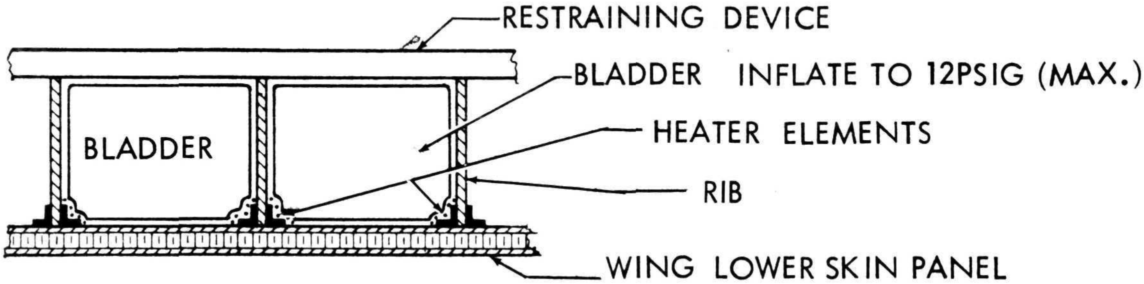
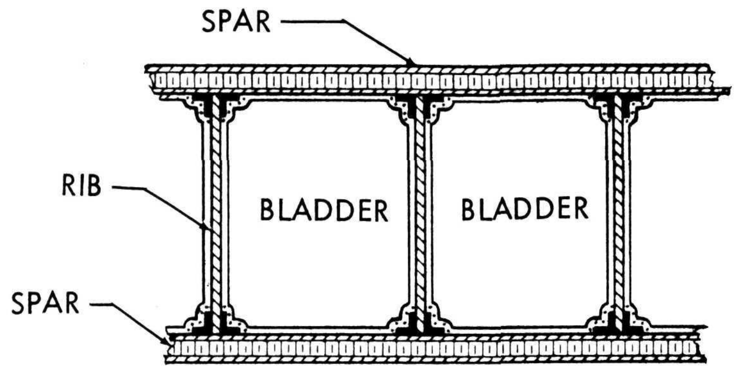


Figure 4.2-1 Bonding Tool Fabrication





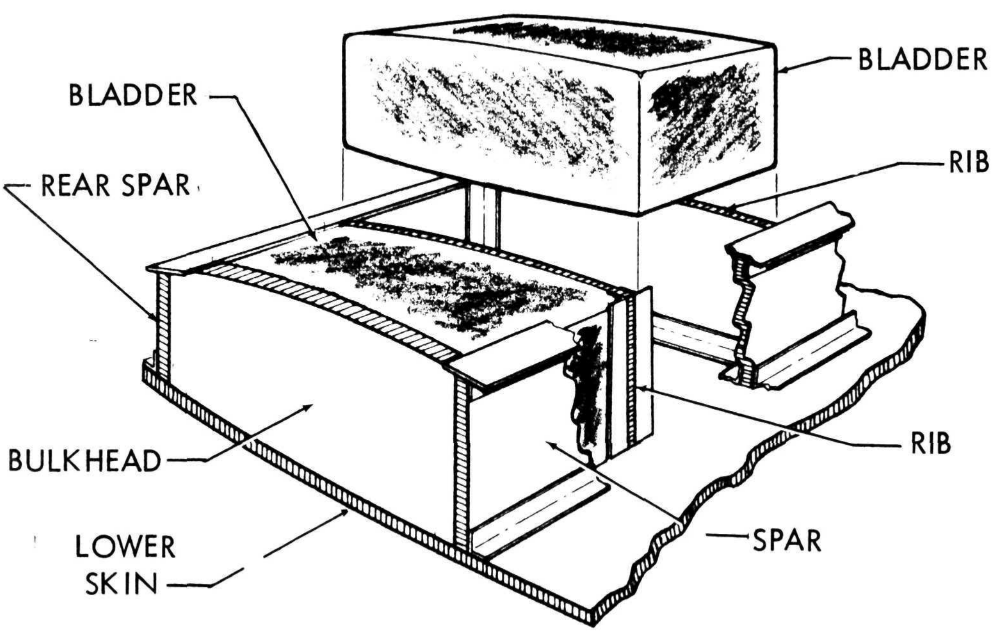


Figure 4.2-2 Tooling Aids for Secondary Bonding

ten manufacturing areas required and a summary of the activities conducted in each are schematically depicted in Figure 4.3-1. More descriptions of activities in each of the manufacturing areas are provided in the following paragraphs.

Area I - Detail Fabrication

The functional setup of this area is based on the types of parts fabricated rather than on aircraft designations. The sub-area functions are as follows:

IA - Core - Blocks of core are foamed (if for the fuselage shell), gang-saw sliced into the required thickness, and spliced into large sections. From these sections, either smaller details are cut or the large sheets are set aside for use in a major component. All core preparation is accomplished in this work area.

IB - Laminated Parts - This is an automated tape laying machine step in which continuous tape or woven cloth materials are made into large panels of the required size, thickness, and fiber orientation. From these large panels, smaller laminate details are cut for further processing. Machines capable of handling all types and widths of materials will be located in this area. Another type of machine-aided fabrication will also be found in this area. The production of stiffeners, angles, and other parts with complex cross sections will be accomplished with a roll shaping process as shown in Figure 4.3-2.

IC - Sandwich Panels - In this section of Area I, large, flat sandwich panels are fabricated and cured. The panels may have any required skin thickness, core thickness, and fiber orientation. The basic steps are: Automatically lay up one skin, apply adhesive (if required) apply core of required thickness and density, apply adhesive (if required), lay-up second skin, bag for vacuum pressure, and cure. Sandwich panels so made may then be cut up to make any of the many sandwich parts required in the airframe.

Area II - Sub-Component Fabrication

Sub-components such as wing spars, cargo floor panels, bulkheads, and many others are fabricated in this area. As an example of the simplified manufacturing procedure sought

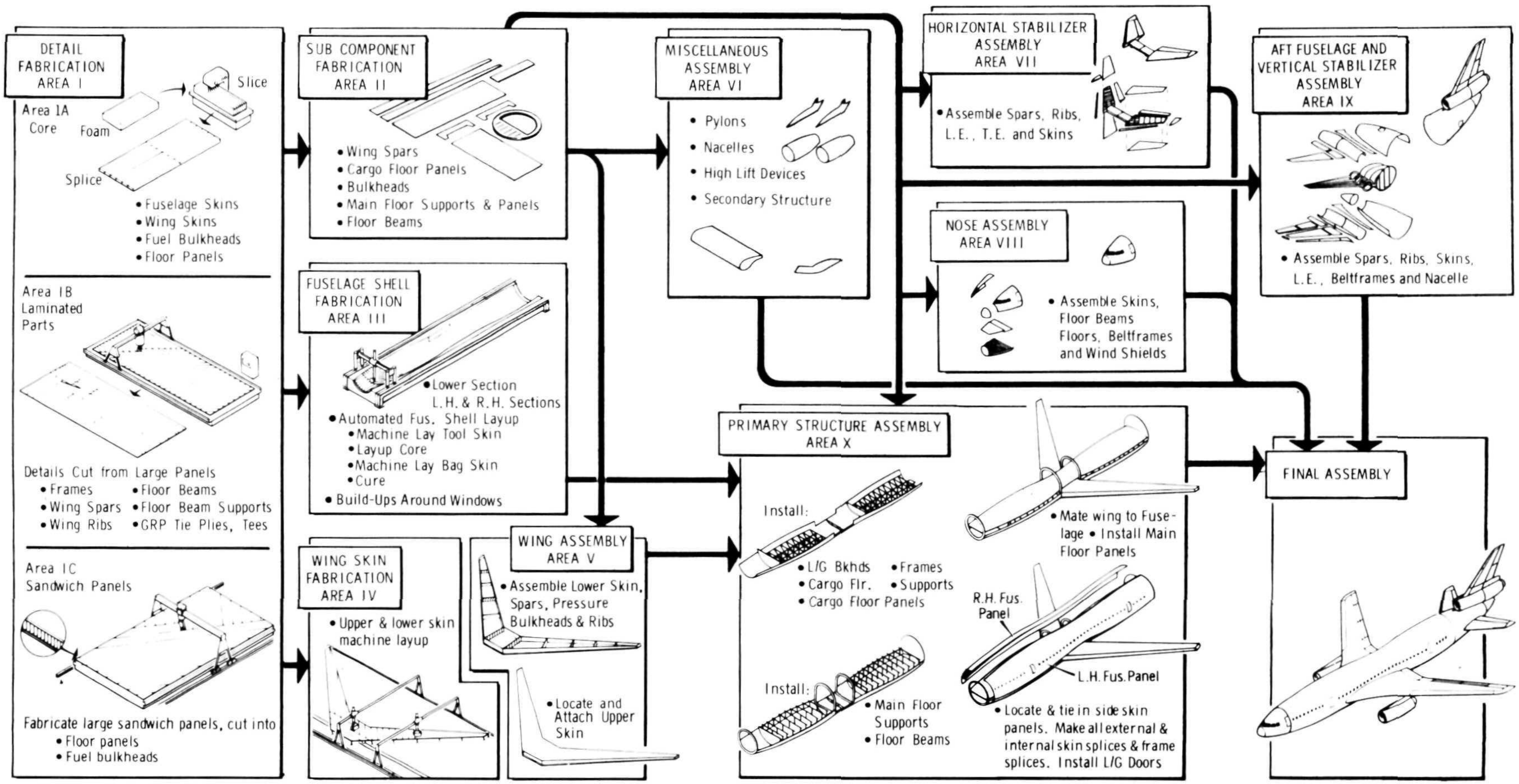


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Figure 4.3-1 Fabrication Approach

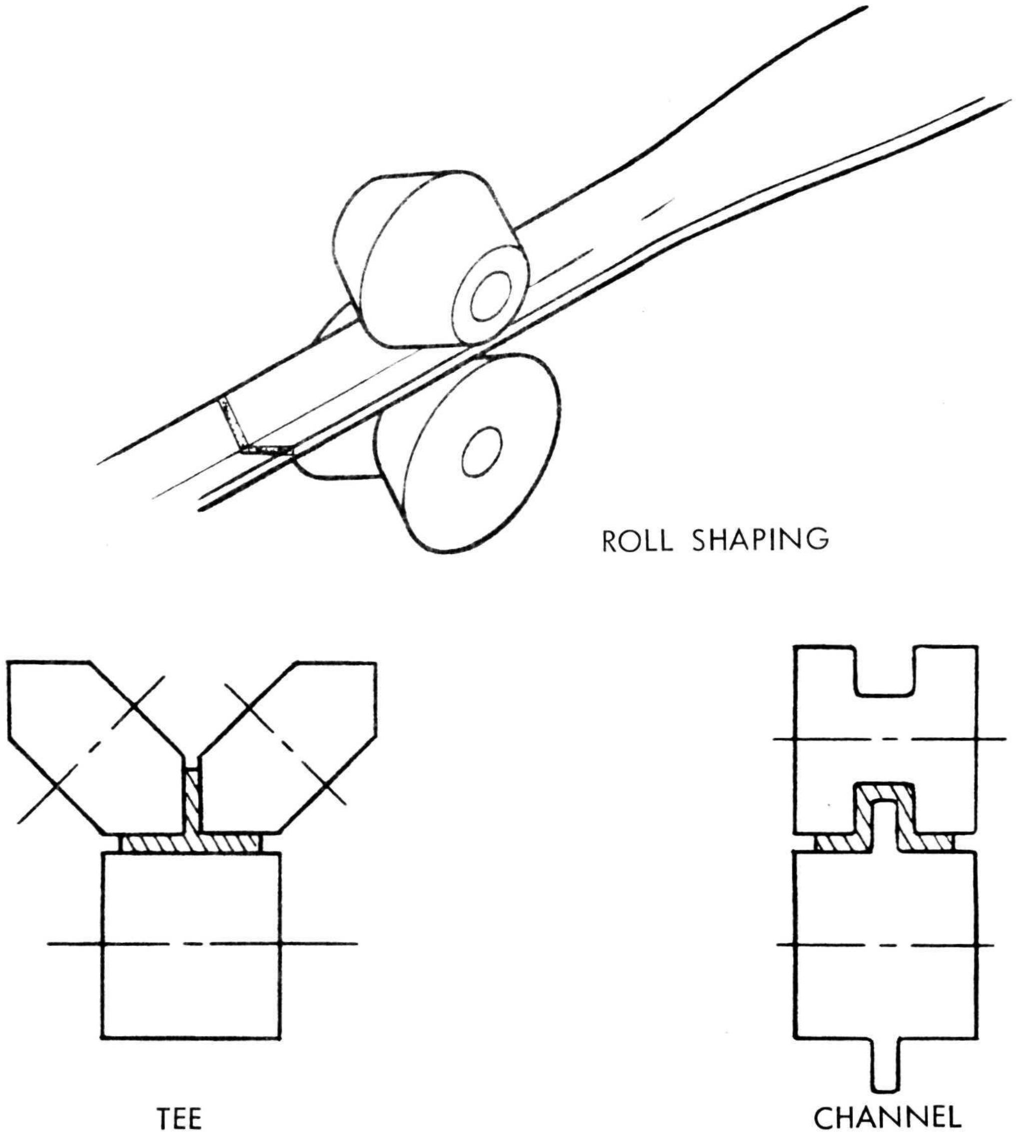


Figure 4.3-2 Basic Roll Shaping Concept

for the parts, consider the following steps required for the production of a core stiffened, back-to-back channel spar:

- o Place a spar channel lay-up (from Area IB) on a male tool. Press down flanges and cure.
- o Apply adhesive.
- o Install core (from Area IA).
- o Apply adhesive.
- o Install side plates on tool and locate a second lay-up in the tool with flanges turned up.
- o Bag for vacuum pressure and cure.

The process utilizes only one basic tool to fabricate and bond a multi-component sub-assembly. This and other simplified procedures will be used to fabricate other components for this airframe.

Area III - Fuselage Shell Fabrication

The fuselage shell is constructed of three sections. The three molds required will constitute a fabrication unit complete with mechanized lay-up equipment and heat source for curing as shown in Figure 4.3-3. A typical lay-up sequence is as follows:

- o Lay-up outer skin.
- o Apply adhesive.
- o Position core as required and apply core splicing adhesive.
- o Apply adhesive.
- o Lay-up inner skin.
- o Bag for vacuum pressure and cure.

Local build-ups in either skin will have been pre-made in Area IB and will be installed here as required during the skin lay-up process.

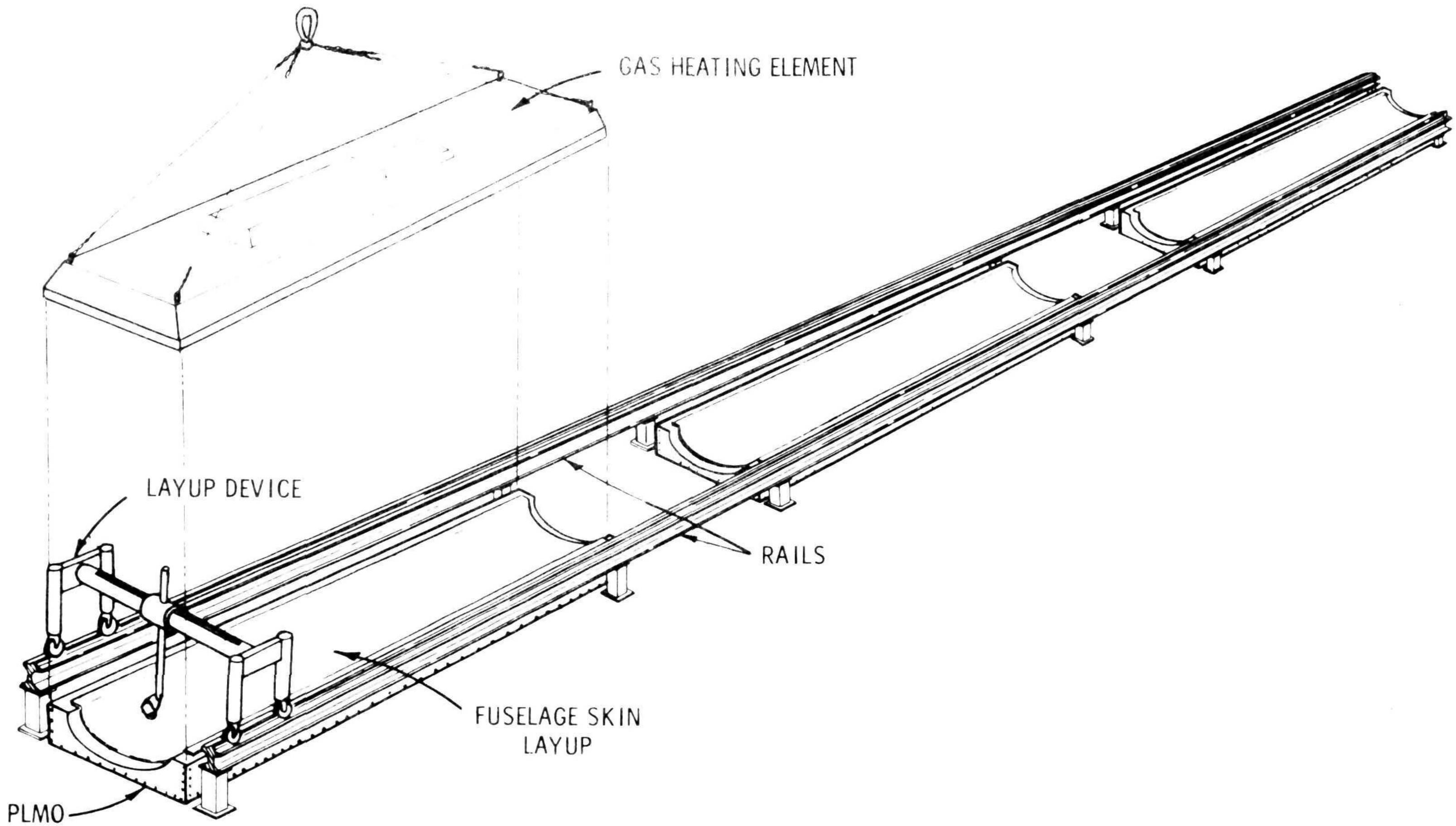


Figure 4.3-3 Fuselage Shell Lay-up and Cure

Area IV - Wing Skin Fabrication

Work in this area is much the same as in Area III and the wing skin lay-up sequence is identical. Again, any local build-ups will have been pre-made and installed in the course of the skin lay-up process. Figure 4.3-4 shows a schematic representation of the mechanized tape laying procedure.

Area V - Wing Box Assembly

The wing assembly will be accomplished in an assembly fixture. The lower skin panel is placed in the fixture. Spars, ribs, pressure bulkheads and other components are then located and holes drilled for mechanical fasteners. All ribs and pressure bulkheads will be attached to the lower skin panel using glass tie plies cured in place. All clips tying ribs and bulkheads to the spars will be installed and bonded to the ribs. Attach holes to the spars will be drilled and blind fastener nuts installed. The upper skin panel, with all rib and bulkhead attach angles bonded to the panel will be located on the lower skin, spar, rib assembly. All attach holes will be drilled and blind fastener nuts installed. Front and rear spars will be removed and mechanical fasteners attaching ribs and bulkheads to upper panel attach angles will be installed. Spars will be re-installed and all mechanical fasteners installed to complete the assembly.

These assembly operations will require extensive handling of the very large parts. One potential method for handling the wing skin panels is to lift and transport them with multiple head vacuum chucks attached to the overhead hoist as shown in Figure 4.3-5.

Area VI - Miscellaneous Assembly

Detail parts and sub-components fabricated in Areas I and II will be used in Area VI to assemble the pylons, nacelles, wing high lift devices and all of the secondary structure required for the airframe.

Area VII - Horizontal Stabilizer Assembly

Assembly of the horizontal stabilizer will follow essentially the same basic procedures as used in assembling the wing. The horizontal stabilizer is divided at Buttock Line 0 of the airframe to facilitate installation.

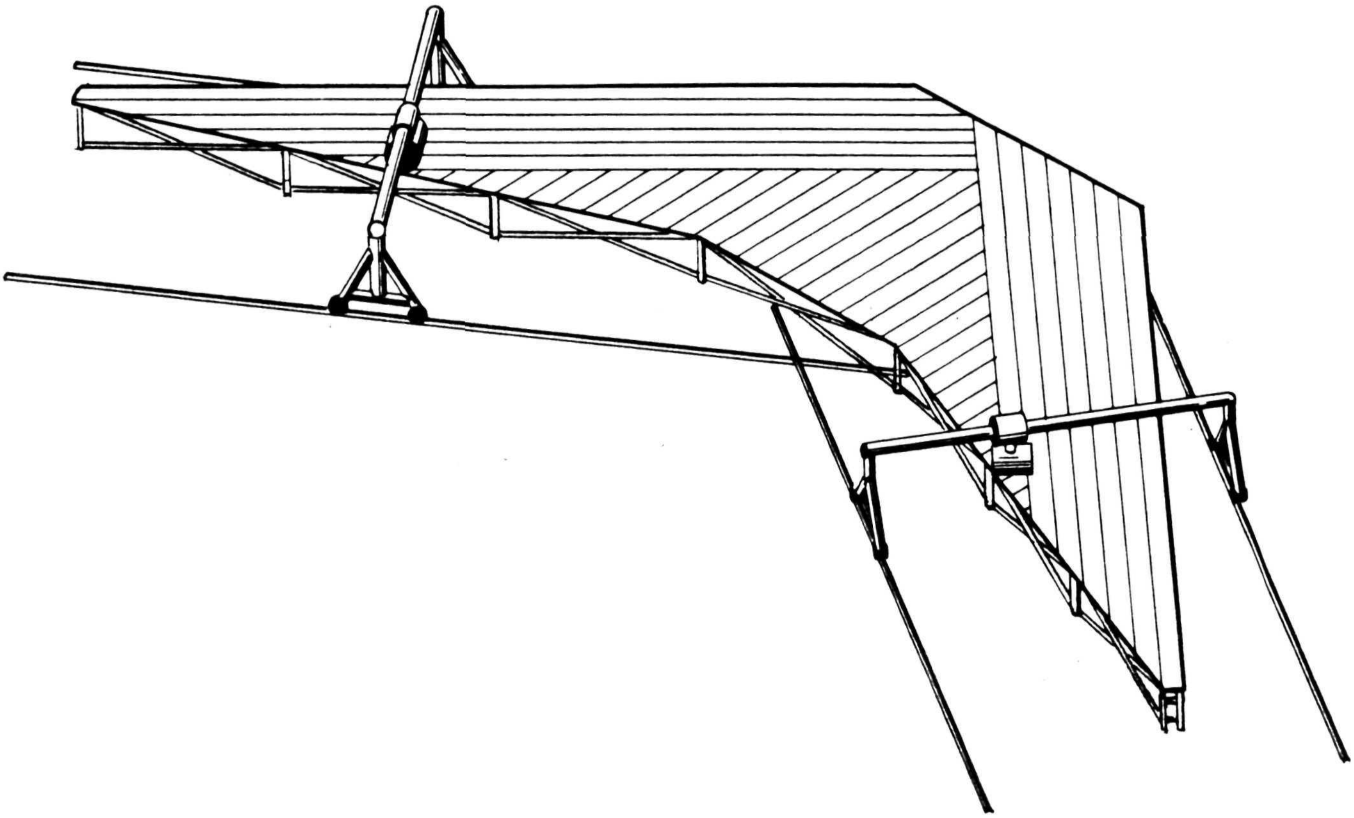


Figure 4.3-4 Wing Skin Lay-up Procedure

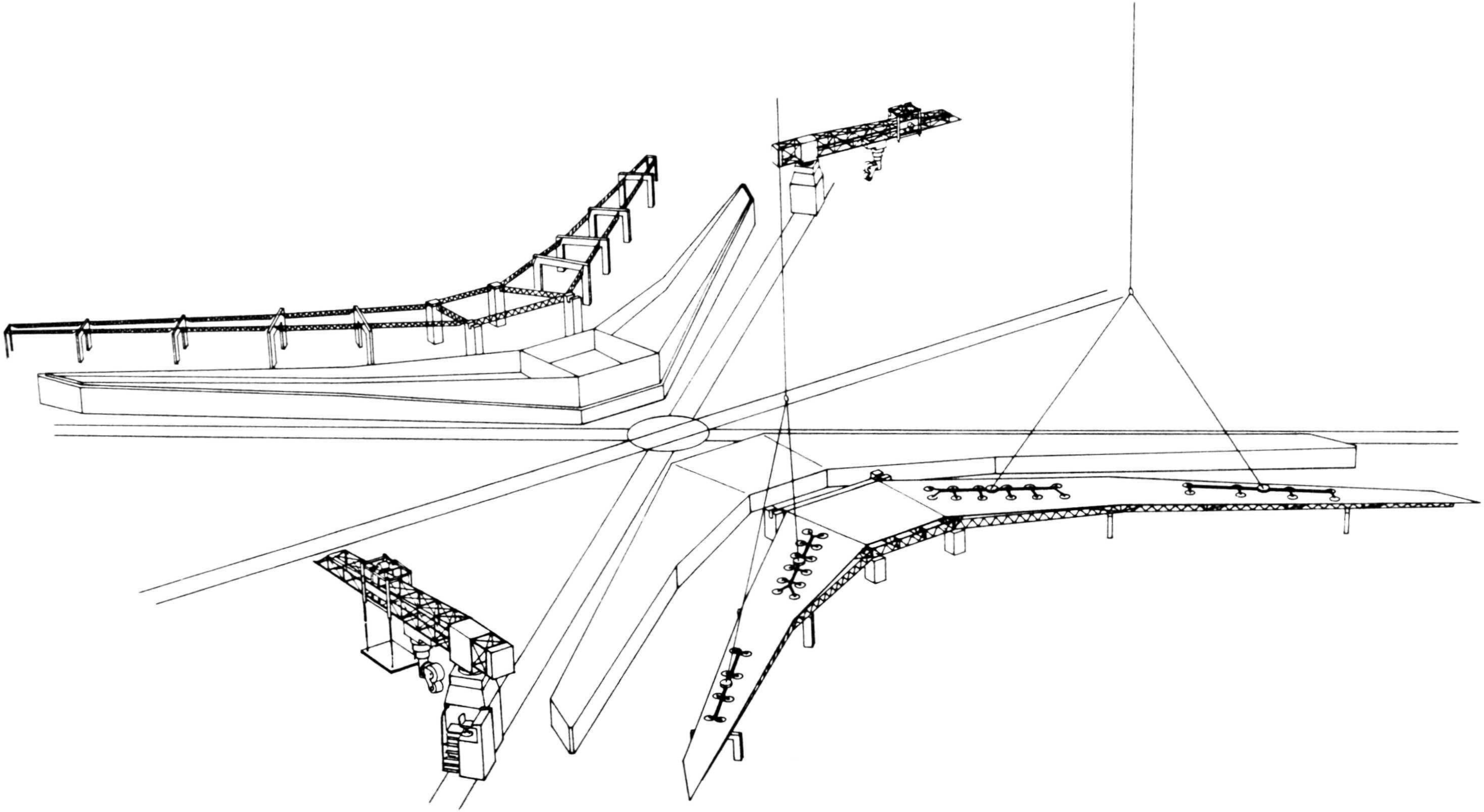


Figure 4.3-5 Wing Skin Handling Procedure

Area VIII - Nose Assembly

Fabrication and assembly of the crew compartment section of the airframe fuselage will closely follow the procedures used in the manufacture of the main fuselage.

Area IX - Aft Fuselage & Vertical Stabilizer Assembly

Normally these would be treated as two separate assembly operations; however, in this design concept the four vertical tail spars extend into the aft fuselage as bulkheads. This characteristic necessitates treating this area as a single major assembly.

The vertical tail skin panels, the spars, ribs and other details will be fabricated using tooling and procedures as outlined in preceding sections. The same is true of the aft portion of the fuselage. These components will be mated by placing the left hand aft fuselage skin panel assembly in an assembly fixture. The vertical stabilizer assembly is mated and the spar-bulkheads attached. The right hand aft fuselage skin is located and all bonds and mechanical fasteners installed. The nacelle covers, inlet duct and rudders will then be installed.

Area X - Primary Structure Assembly

The lower fuselage skin panel will be located in an assembly fixture and all frames and bulkheads located and installed. Next installed will be the cargo floor beams, supports, and floor panels. The wing box assembly is then mated to the fuselage. Left and right-hand upper skin panels are located and the splice plates installed. All ring frame splices will be made and the main floor beams, supports, floors, and seat tracks installed. Forward and aft pressure bulkheads will be installed. All landing gears and landing gear doors will be installed.

At this point in the manufacture of the airframe all of the primary and secondary assemblies have been completed and final assembly is the only remaining activity.

4.4 FINAL ASSEMBLY

Final assembly of the composite transport airframe will utilize major assembly fixtures and assembly techniques which are identical to those used for conventional metal aircraft.

SECTION 5

COST ANALYSIS

As a part of the ATT systems studies sponsored by NASA, General Dynamics identified the potential of significant economic benefits for commercial transport aircraft which contain major structural applications of advanced composite materials (Reference 2-1). These studies showed that the aircraft could cost less to produce and to operate. In the present studies documented in this report, certain manufacturing costs of the aircraft were evaluated with an alternate cost estimating approach to provide more detail and additional insight into the earlier findings.

5.1 INITIAL STUDIES

Those elements of aircraft structure manufacturing cost which received attention in this effort are shown in Tables 5.1-1 and 5.1-2 along with the corresponding costs in terms of manhours from the ATT system study conducted at General Dynamics. The costs discussed in Reference 2-1 are presented in terms of 1970 dollars and, while the manhour costs shown in Tables 5.1-1 and 5.1-2 do not appear in Reference 2-1, they are a part of the basic information which produced the dollar estimates.

The anticipated tooling costs presented in Table 5.1-1 were prepared using cost estimating equations which have been developed from historical data representing both military and commercial aircraft. Some of the important features or assumptions incorporated in the use of these equations include the following:

(1) tool engineering hours are 35 percent of tool manufacturing hours,

(2) tooling quality control hours are 7.5 percent of tool manufacturing hours,

(3) costs of rate tooling to permit six per month production of aircraft are 15 percent of the basic tooling costs,

(4) tool maintenance rate is 0.9 percent per delivery month of non-recurring tooling costs, and

TABLE 5.1-1

TOOLING COST FROM SYSTEM STUDY*

Component	Manhours (Millions)				
	Basic**	Rate***	Total Non-Recurring	Maintenance	Total
Total Airframe Structure	3.858	.579	4.437	1.997	6.434

*Initial Studies (Reference 2-1).

**Basic: Non-recurring tooling required to make one airframe.

***Rate: Additional tooling to make six airframes per month.

Table 5.1-2

FACTORY COSTS FROM SYSTEM STUDY

Component	First Unit, Manhours (millions)	Total 250 Units, Manhours (millions)
Wing	0.214	12.463
Tail	0.077	5.698
Fuselage	0.247	14.186
Nacelles	0.072	5.780
Total	0.610	38.127

(5) the delivery period for the 250 aircraft is 50 months.

Existing cost estimating equations based upon historical data do not adequately account for fabrication with advanced composite materials. The anticipated factory costs presented in Table 5.1-2 were prepared using data developed in studies which compared expected costs of conventional metal aircraft to those of advanced composite aircraft. The factory costs shown in Table 5.1-2 include 15 percent for associated quality control functions.

The data in Tables 5.1-1 and 5.1-2 show that the ATT system study at General Dynamics estimated that approximately 45 million manhours would be required to manufacture the structure for the 250 aircraft and that 86 percent of the manhour expenditures would be in the factory.

5.2 PRESENT STUDIES

The present studies documented in this report provided additional detail for the tooling and factory tasks and permitted alternate approaches to be used in estimating tooling and factory costs.

5.2.1 Tooling Costs

The anticipated tooling costs were prepared in detail for each component in accordance with the manufacturing approach described in Section 4. Tool manufacturing manhour estimates were developed for all tools envisioned for each component including tool maintenance and then tool engineering manhours were estimated at 30 and 40 percent of tool manufacturing manhours for non-recurring and recurring costs, respectively. Quality control manhours were estimated at 7.5 percent of tool manufacturing manhours. The resulting expected tooling costs are summarized in Table 5.2-1.

There are some significant variances between the data that appear in Tables 5.1-1 and 5.2-1 which warrant discussion. The costs of basic tooling which is the non-recurring tooling required to build the first unit are estimated at 2.390 million manhours, which is only 62 percent of the 3.858 million manhours estimated during the initial studies

Table 5.2-1

EXPECTED TOOLING COSTS

Component	Manhours (Millions)				
	Basic	Rate	Total Non-Recurring	Maintenance	Total
Fuselage	0.966	1.896	2.862	1.823	4.685
Wing	0.879	1.288	2.167	1.387	3.554
Horizontal Stabilizer	0.100	0.158	0.258	0.165	0.423
Aft Fuselage and Vertical Tail	0.445	1.039	1.484	0.942	2.426
Total	2.390	4.381	6.771	4.317	11.088

(Reference 2-1). This relationship is entirely consistent with other comparisons of tooling required for metal versus advanced composite airframe structure conducted at General Dynamics. In general, these comparisons indicate that for one of a kind, the tooling required for the composite airframe will be 60-70 percent of the cost of that required for the conventional metal airframe. Because the original estimate shown in Table 5.1-1 was based upon historical data generated for conventional metal aircraft, the 62 percent relationship between the two estimates might have been anticipated. This relationship is explained by the fact that although the tools required for the advanced composite airframe are more complicated, there are so far fewer of them required to fabricate the larger pieces of structure that the total tooling cost is estimated to be reduced.

On the other hand the total non-recurring costs of tooling are estimated to be 53 percent greater than it was in the initial studies (Reference 2-1), 6.771 million manhours versus 4.437 million manhours. This variance is also related to the differences between tooling for metal versus tooling for composite aircraft. Conventional metal aircraft may have hundreds of detail parts and consequently hundreds of tools for each major assembly. Large quantities of metal detail parts can be produced to meet six aircraft per month production requirements with little or no increase in tool quantities. The only tools which must be augmented are those in which parts or assemblies must remain in station for long enough periods of time to require duplicate tools. The costs of these few tools is small when compared to the total basic tooling costs (about 15 percent as discussed in Section 5.1). The very large component assemblies and the nature of the manufacturing processes anticipated for the advanced composite aircraft require very large, complex bonding and assembly tools, but relatively few detail parts and tools. It is these complex assembly tools which must be augmented in order to meet the required production rate and hence the 53 percent greater costs.

The approach used in the present studies to estimate tool maintenance costs was a percentage of the total non-recurring tool costs as it was for the initial studies. The variance between maintenance or recurring costs in the two studies is affected by the larger non-recurring costs in the later study and the assumption that the total production program will be 69 months long as shown in Figure 4.1-1 instead of 50 months as discussed previously.

The total result of these variances is an expected total tooling cost which is 72 percent greater than originally estimated.

5.2.2 Factory Costs

The anticipated factory costs were prepared by considering the detail weight breakdown of each component by material type and historical data related to factory costs experienced with these material types. The detail weight statement of each component is shown in Table 5.2-2. Consideration of historical cost data provides a credible base for estimating the factory costs. The following paragraphs describe the sources of the background cost information and then the selections utilized in this study.

5.2.2.1 Historical Cost Data

Historical cost data for advanced composite applications were evaluated from the following components:

F-111 Aft Fuselage - This component, shown in Figure 5.2-1, represented the F-111 aft centerbody section between the engines and was fabricated primarily of graphite-epoxy materials by General Dynamics. Hand lay-up techniques were used to fabricate the parts and there was extensive use of bonded sandwich structure with some stiffened-sheet and chopped fiber molded structure also used (References 5-1 and 5-2).

A-4 Stabilizer - McDonnell Douglas fabricated an A-4 horizontal stabilizer primarily of graphite-epoxy materials. Hand lay-up techniques were used to fabricate the parts and the surface panels were of plate construction. The substructure utilized bonded sandwich construction (References 5-3 and 5-4).

F-5 Confirmation Component - General Dynamics fabricated this 10-foot (3.05M) long component, shown in Figure 5.2-2, which represented a section of the F-5 aircraft. Graphite-epoxy materials were used primarily and hand lay-up techniques were used to fabricate the unstiffened plate construction used extensively in the shell structure and substructure (References 5-5 and 5-6).

F-5 Shell Tool - General Dynamics fabricated this 20-foot (6.1M) long tool in support of the advanced composite

Table 5.2-2

DETAIL WEIGHT STATEMENT

Component Material	Weight lb	Weight Kg
Wing		
Composites		
Covers	6,200	2,812
Spars	2,600	1,179
Other	9,000	4,082
Total	17,800	8,073
Core and Adhesive	3,100	1,406
Metal	5,400	2,449
Total	26,300	11,928
Tails		
Composites		
Covers	1,250	567
Other	2,390	1,084
Total	3,640	1,651
Core and Adhesive	870	395
Metal	640	290
Total	5,150	2,336
Fuselage		
Composites		
Shell	8,920	4,045
Other	10,867	4,928
Total	19,787	8,973
Core and Adhesive	2,448	1,110
Metal	5,840	2,649
Total	28,075	12,732
Nacelles		
Composites	2,270	1,029
Core and Adhesive	573	260
Metal	1,627	738
Total	4,470	2,027
Total	63,995	29,023

Figure 5.2-1 F-111 Aft Fuselage Component

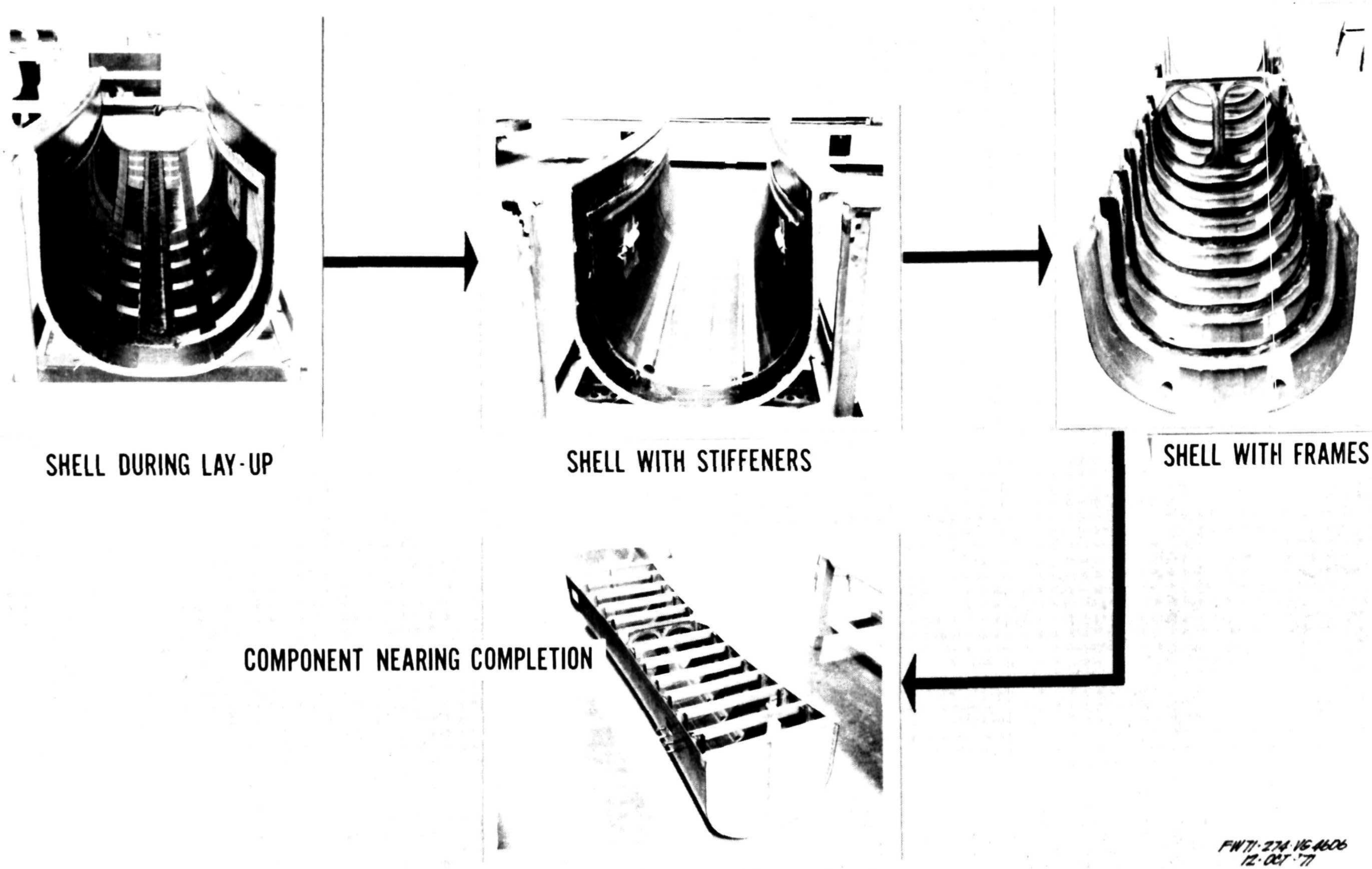


Figure 5.2-2 F-5 Confirmation Component

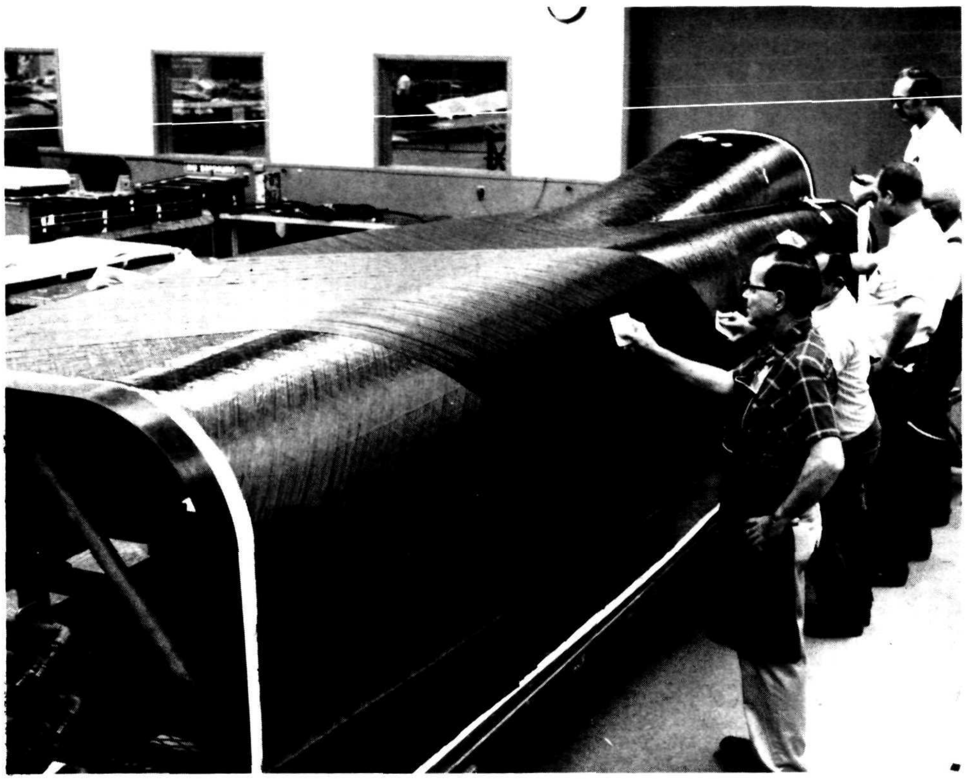
F-5 fuselage fabrication. The tool, shown in Figure 5.2-3, was fabricated with graphite-epoxy skins of constant ply count bonded to a constant thickness aluminum core. Hand lay-up techniques were used to fabricate the tool (Reference 5-6).

F-5 Fuselage - This component, shown in Figure 5.2-4, is a mid-fuselage section of the F-5 aircraft and was fabricated by General Dynamics primarily of graphite-epoxy materials. Hand lay-up techniques were used to fabricate details. The shell structure is unstiffened plate with integral longerons and build-ups and the substructure contains unstiffened plate with some sandwich construction (References 5-5 and 5-6).

The relationships of factory manhour costs, excluding quality control and all other support functions, and structural weight produced are shown for these components and for their shell or cover structure in Figures 5.2-5 and 5.2-6, respectively. The radial lines emanating at the origins of these two figures are reference lines of constant cost per unit of structural weight produced. There are several important observations to be made from these two figures.

Figure 5.2-5 shows that the F-5 fuselage component factory cost was more than twice that of the F-111 fuselage component even though they represented approximately the same structural weight. This cost difference was created by the extremely more complex design approach used for the F-5 component. This figure also shows that the industry has already fabricated prototype components at approximately 16 manhours per pound (35.3 manhours per kilogram) even though it is still in primitive stages of production-oriented material, design concept, and manufacturing process development.

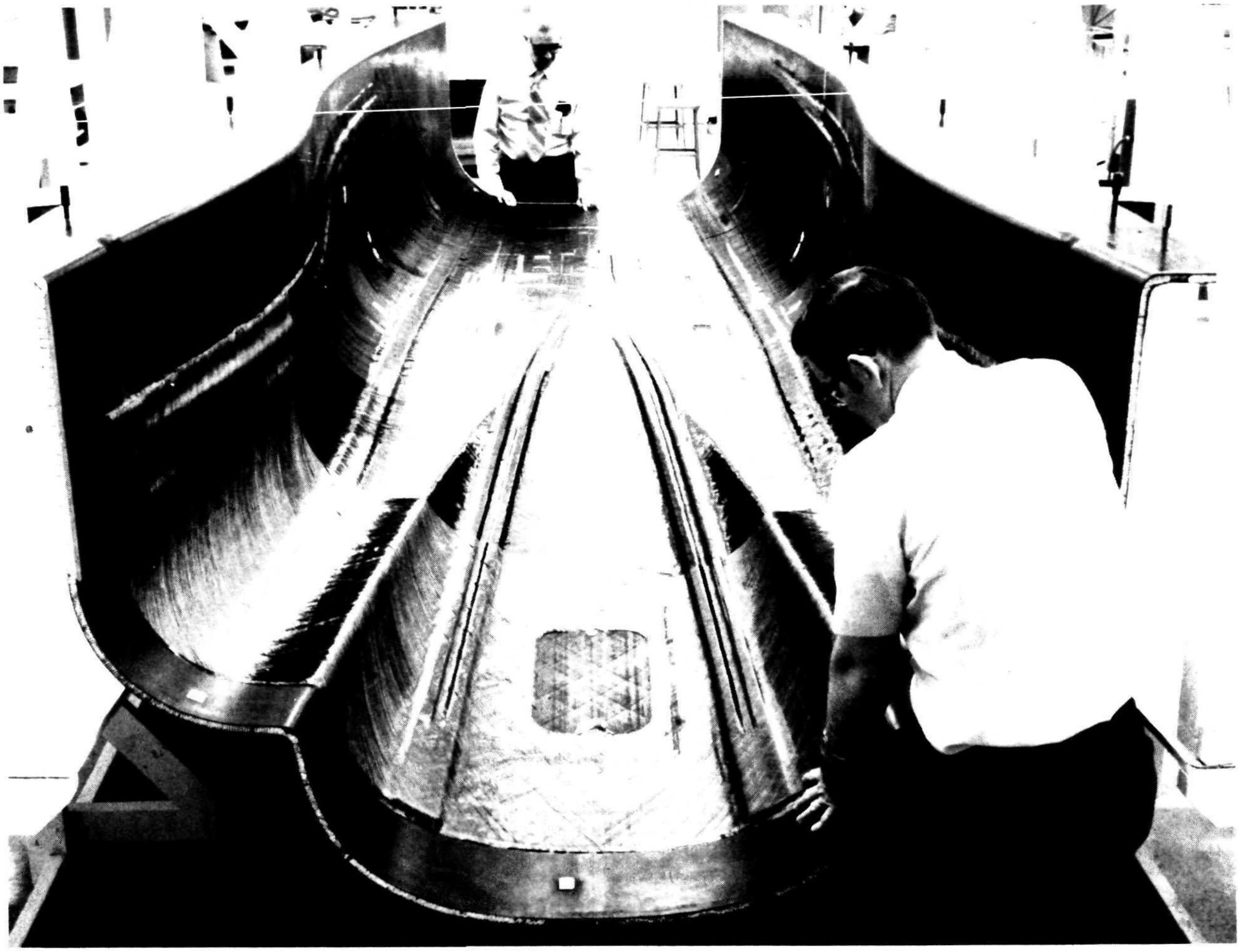
Figure 5.2-6 shows even greater cost differentials between simple and complex structure. The F-5 fuselage shell and the F-5 shell tool skins have exactly the same external shape but the tool skins cost less than one-fourth of the shell cost even though it weighed more than one-fourth more. The tool skins cost less because they were of constant ply count and therefore much less complex than the shell with its integral buildups for longerons, frame and bulkhead flanges, and cutout reinforcement.



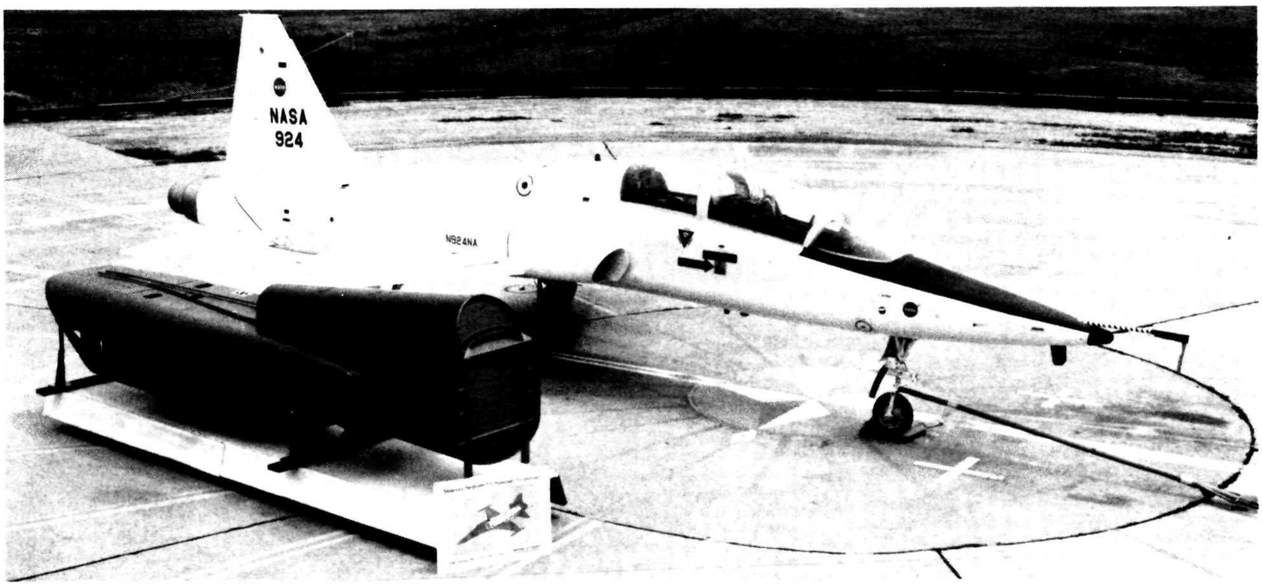
(a) INNER SKIN LAYUP

(b) COMPLETED TOOL

Figure 5.2-3 F-5 Shell Bonding Tool



(a) UPPER SHELL LAYUP



(b) COMPLETED COMPONENT

Figure 5.2-4 Advanced Composite F-5 Fuselage

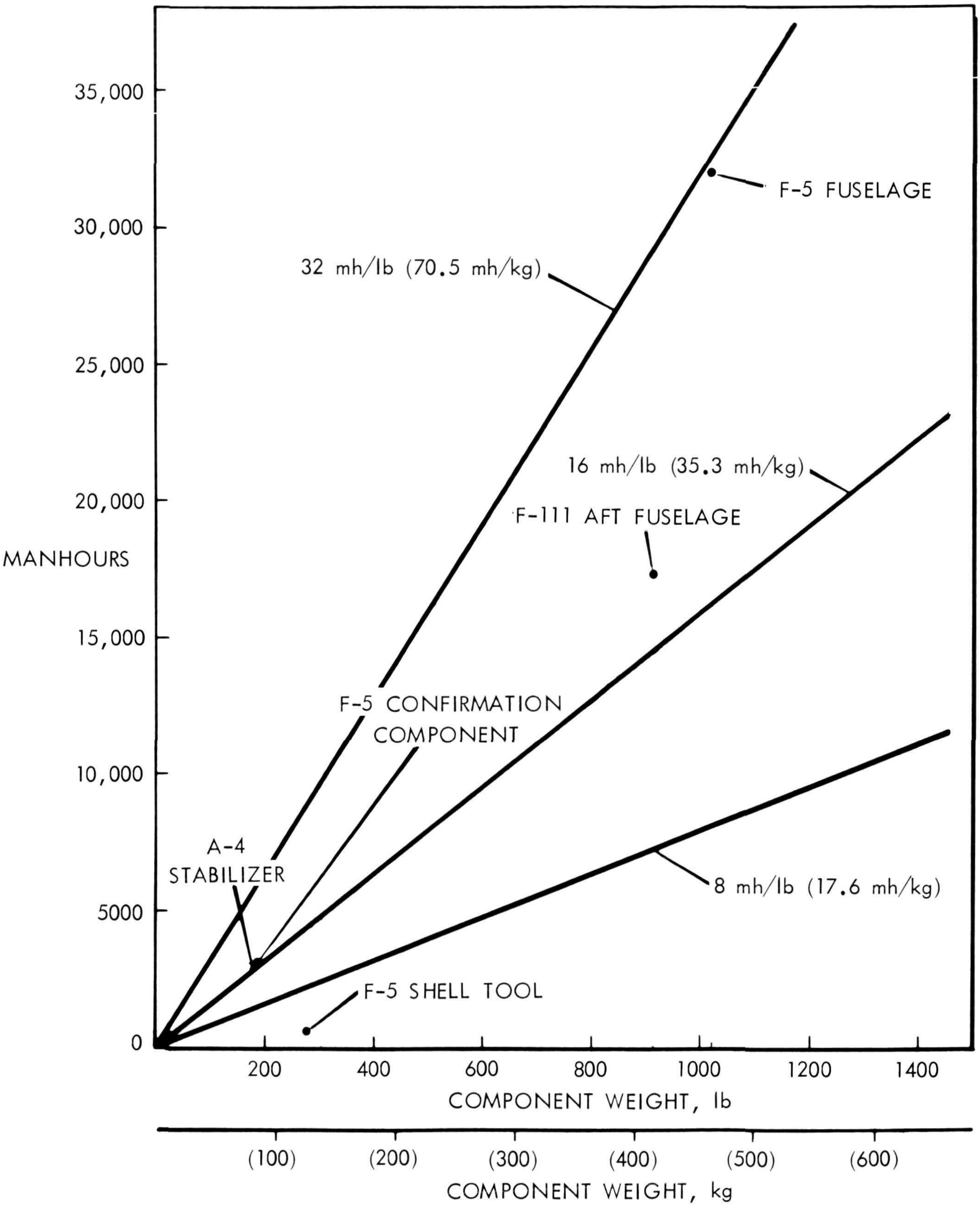


Figure 5.2-5 Total Component Cost-Weight Relationships

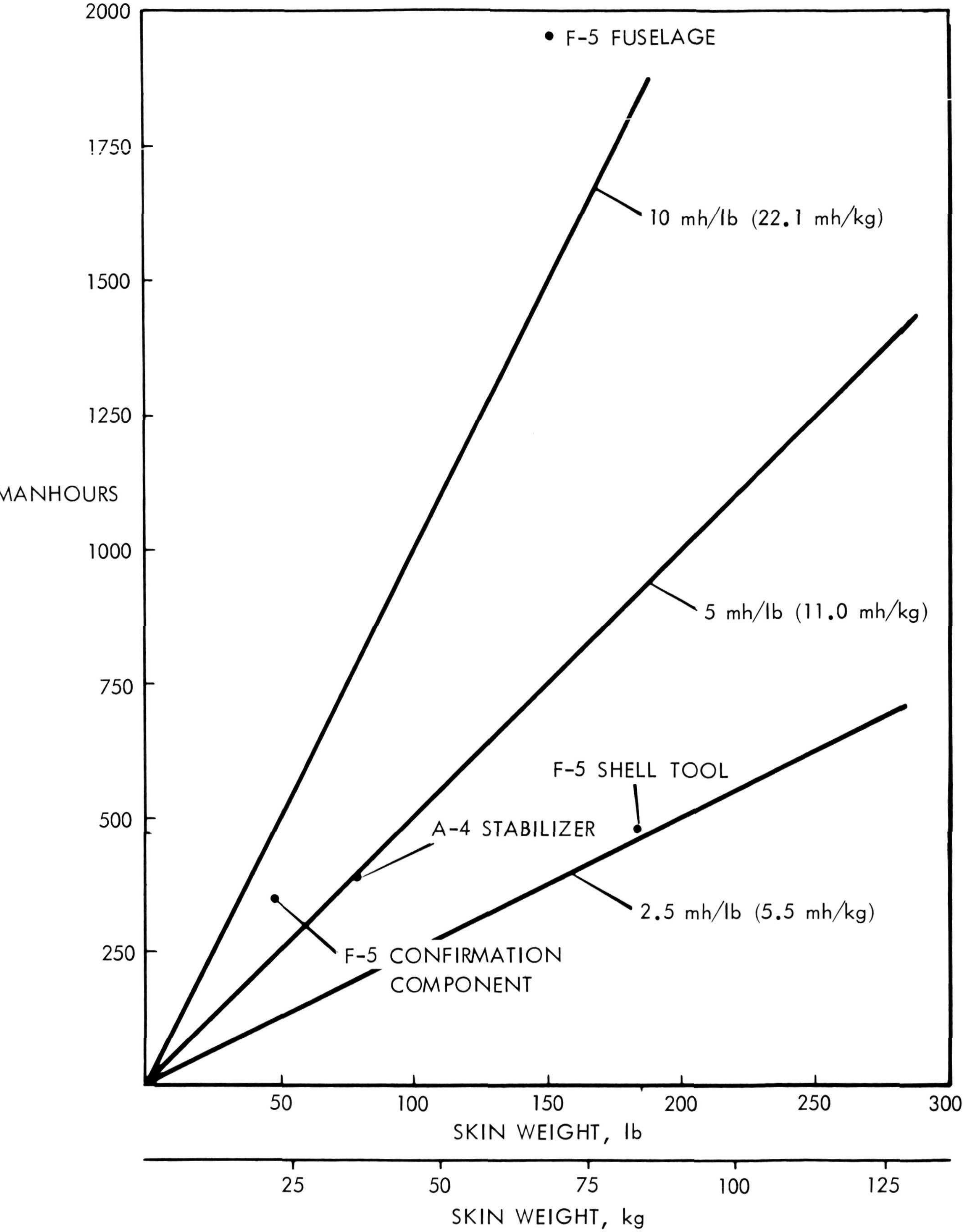


Figure 5.2-6 Surface Structure Cost-Weight Relationships

None of the above-mentioned components have been committed to production and there is very little historical cost data available on production learning curves for advanced composite materials. Composites Recast recognized an extensive amount of production data available for fiberglass composites and resulting learning curves in the 80 to 85 percent range (Reference 5-7). Much of this fiberglass production incorporates at least semi-automated fabrication processes as does the manufacturing approach anticipated for the transport. Two boron-epoxy components--the F-4 rudder and the F-14 horizontal stabilizer--have been fabricated under production conditions. Both components were fabricated with hand lay-up techniques and both developed approximately 85 percent learning curves. Conventional metal aircraft structure have developed learning curves in the 73 to 80 percent range.

Historical cost data for application of core and adhesives were evaluated from the F-5 shell tool discussed previously. The aluminum core was used in large sections and constant thickness as it will be for much of the transport structure. It also experienced cleaning operations and adhesive application much like those anticipated for the transport structure. The core and adhesive weighed 84 pounds (38.1 kg) and cost 212 manhours to complete the required manufacturing operations. The cost per pound was 2.52 manhours per pound (5.57 manhours per kilogram).

Historical cost data for the metal structure were evaluated from the DC-10 fuselage experience. The fuselage section fabricated at General Dynamics weighs approximately 37,400 pounds (16,960 kg) and the first unit's factory cost was approximately 479,400 manhours, or approximately 12.8 manhours per pound (28.3 manhours per kilogram).

5.2.2.2 Estimated Costs

Consideration was given to the historical cost data presented in Section 5.2.2.1 in estimating the factory costs associated with the transport aircraft at the first unit and for the total production program. It should be noted that because the components expected to be produced for this aircraft are much larger than those which have thus far been produced with composite materials and the manufacturing approach assumes the use of manufacturing equipment which has not yet been developed, the cost estimates related to composite applications must be considered as extrapolations of the historical data.

The basic questions in using historical data for estimating the unit costs of applying the composite materials to the wing, tails, and fuselage cover structure and wing spars are concerned with anticipating the penalties associated with the increased sizes of parts and the benefits associated with the use of automated lay-up equipment. Each sandwich facing in the wing and fuselage has approximately an order of magnitude more material than the F-5 shell tool skins contained; however, at normal operating speeds of 400 inches (10.2M) per minute the tape laying machines described in Section 4 can place material at the rate of 0.006 hours per pound (0.013 hours per kilogram). The unit factory cost including quality control support estimated for the composite materials in the cover structure of the wing, tails, and fuselage was 5.0 manhours per pound (11.0 manhours per kilogram) of material at the first unit. This unit cost is more than twice as expensive as the F-5 shell tool skins were and almost two orders of magnitude more expensive than using the tape laying machine to place the material; however, the human inefficiencies associated with handling such large structure appear to warrant these increases.

The unit factory cost including quality control estimated for the composite materials in the wing spars at the first unit was 7.5 manhours per pound (16.5 manhours per kilogram) of material. Although these pieces of structure are smaller than the wing and fuselage covers and the associated inefficiencies should decrease, the spars will be more complex with the required reinforcements for doors, cutouts, and local load introduction points. These increased design and manufacturing complexities are expected to add the 50 percent to the unit cost when compared to the rates used for the wing and fuselage covers.

The unit factory cost including quality control for all other composite details on the first article produced was estimated to be 10.0 manhours per pound (22.1 manhours per kilogram). Most of the parts in this category are small when compared to the wing and fuselage covers; however, most of them will also be much more complicated to fabricate because they will contain integral ply build-ups, drastic changes in shape, joints and attachments, and other complex features which will increase the unit fabrication cost.

The unit factory cost including quality control for the core and adhesive on the first article produced was estimated

Table 5.2-3

EXPECTED FACTORY COSTS

Component Material	Weight		First Unit			Total 250 Units
	lb	Kg	mh/lb	mh/kg	Total Manhours (Millions)	Total Manhours (Millions)
Wing						
Composites						
Covers	6,200	2,812	5.0	11.0	0.031	
Spars	2,600	1,179	7.5	16.5	0.019	
Other	9,000	4,082	10.0	22.1	0.090	
Total	17,800	8,073			0.140	
Core and Adhesive	3,100	1,406	10.0	22.1	0.031	
Metal	5,400	2,449	14.7	32.4	0.079	
Total	26,300	11,928			0.250	15.830
Tails						
Composites						
Covers	1,250	567	5.0	11.0	0.006	
Other	2,390	1,084	10.0	22.1	0.024	
Total	3,640	1,651			0.030	
Core and Adhesive	870	395	10.0	22.1	0.009	
Metal	640	290	14.7	32.4	0.009	
Total	5,150	2,336			0.048	3.039
Fuselage						
Composites						
Shell	8,920	4,045	5.0	11.0	0.045	
Other	10,867	4,928	10.0	22.1	0.109	
Total	19,787	8,973			0.154	
Core and Adhesive	2,448	1,110	10.0	22.1	0.024	
Metal	5,840	2,649	14.7	32.4	0.086	
Total	28,075	12,732			0.264	16.716
Nacelles						
Composites	2,270	1,029	10.0	22.1	0.023	
Core and Adhesive	573	260	10.0	22.1	0.006	
Metal	1,627	738	14.7	32.4	0.024	
Total	4,470	2,027			0.053	3.356
Total	63,995	29,023			0.615	38.941

to be 10.0 manhours per pound (22.1 manhours per kilogram). This rate is approximately four times that experienced on the F-5 shell tool and some of the details on the transport should be completed for 2.5 manhours per pound (5.5 manhours per kilogram); however, there will be many pieces of core and adhesive used which will cost much more than that. Foaming the core for the fuselage shell will add some cost. In addition, much of the core other than that in the cover structure and the spars will have to be machined locally and prefitted which adds rapidly to the cost.

The unit factory cost including quality control for the metal parts on the first article produced was estimated to be 14.7 manhours per pound (32.4 manhours per kilogram) which was that factory cost experienced on the DC-10 with the addition of 15 percent for quality control.

The factory learning curve assumed to be applicable for each of the four major components reflecting the specified manufacturing processes and production rate is an 80 percent curve breaking to an 87 percent curve at Ship 100. Use of this curve assumes rapid learning and process improvement during the first 100 units but then reduced cost improvements for the remaining 150 units. This learning curve is not as steep as some experienced for metal structure but the required composite manufacturing processes may never permit such steep learning curves to be realized.

The expected costs discussed in the previous paragraphs have been applied to the weights shown earlier in Table 5.2-2 and the resulting expected factory costs for the first unit produced are shown in Table 5.2-3. The expected total cost of the 250 unit fabrication program is also shown in Table 5.2-3 in terms of factory plus quality control support manhours.

Comparisons of the data in Tables 5.1-2 and 5.2-3 show that the expected costs of the wing and fuselage as a result of the present study are greater than those which resulted from the system study and the reverse is true for the tails and nacelles. The total expected manhour costs, both for the first unit and the total 250 units, are shown to be in remarkable agreement when the corresponding results of the two complete different estimating approaches are compared.

5.3 COST SUMMARY

A summary of the estimated manhour costs determined during the initial and present studies is presented in Table 5.3-1. This summary shows that the differences in the expected tooling costs discussed in Section 5.2.1 are primarily responsible for the differences which appear in the total expected manhour costs.

Results of the present study have been prepared in terms of dollar costs as shown in Tables 5.3-2 through 5.3-4. Similar results from the initial study are also shown for comparative purposes. The rates used in preparing these costs were assumed to be thirty dollars per pound for composite materials, eight dollars per pound for all other materials, twelve dollars per manhour for factory and quality control labor, and 16.70 dollars per manhour for tooling labor. The expected average cost of the 250 aircraft is 13.98×10^6 dollars as shown in Table 5.3-4.

The difference between the total estimated costs of the 250 aircraft as developed in the two studies is shown in Table 5.3-4 to be 155.71×10^6 dollars. An analysis of variances indicates that 10.3×10^6 dollars of this difference is attributed to the increased factory and quality control direct labor, 67.7×10^6 dollars is attributed to the increased weight of composite materials used in the airplane (43,497 lb. (19,699 kg) instead of 31,180 lb. (14,130 kg)), and 77.71×10^6 dollars is attributed to the increased tooling direct labor.

Table 5.3-1

COST SUMMARY

Study Function	Cost, Manhours (Millions)	
	First Unit	Total 250 Units
Initial		
Tooling	3.858	6.434
Factory	0.610	38.127
Total	4.468	44.561
Present		
Tooling	2.390	11.088
Factory	0.615	38.941
Total	3.005	50.029

Table 5.3-2

FACTORY, QC, AND MATERIALS COST ANALYSIS

	Component Weight		Initial Studies		Present Studies	
	lb	kg	First Unit Cost, Million $	Total 250 Aircraft Cost, Million $	First Unit Cost, Million $	Total 250 Aircraft Cost, Million $
Wing	26,300	11,950	3.06	272.	3.60	340.
Tail	5,150	2,340	1.02	91.	.70	67.
Fuselage	28,075	12,800	3.53	313.	3.83	366.
Landing Gear	12,545	5,680	.98	75.	.98	75.
Flight Controls	4,210	1,920	1.08	111.	1.08	111.
Nacelles	4,470	2,030	.91	81.	.72	62.
Total Structure Wt	80,750	36,600				
Water Injection System	265	120	.08	8.	.08	8.
Fuel System	1,525	691	.37	28.	.37	28.
Engine Controls	195	88	.05	4.	.05	4.
Starting	140	64	.04	4.	.04	4.
Reversers	2,105	960	.46	48.	.46	48.
Total Propulsion Assoc Wt	4,230	1,920				
Hydraulics	1,960	890	.48	49.	.48	49.
Instruments	1,740	790	1.05	133.	1.05	133.
Electrical	3,217	1,460	.84	86.	.84	86.
Furnishings	23,169	10,500	1.73	154.	1.73	154.
Air Conditioning & Anti-Ice	3,990	1,810	.56	50.	.56	50.
Auxiliary Gear	45	20	.01	1.	.01	1.
APU	908	410	.20	21.	.20	21.
Total Subsystem Wt	35,029	16,000				
Subtotal Costs			16.45	1,529	16.78	1,607
Factory Assembly			2.06	136.	2.06	136.
Mission Equip Assembly			.04	3.	.04	3.
Factory Acceptance			.76	50.	.76	50.
Engines (P&W-STF-429)	14,959	6,790	2.51	628.	2.51	628.
Avionics			.91	187.	.91	187.
Total Cost			$22.74	$2,533.	$23.06	$2,611

TABLE 5.3-3

TOTAL NON-RECURRING--RDT&E COSTS

Cost Element	Cost, Million $			
	Initial Studies		Present Studies	
Airframe Design		77.30		77.30
Air Vehicle Integration		27.05		27.05
Development Shop & Material Support		107.20		107.20
Basic Tooling				
Wing, Tail, Fuselage & Nacelles	64.43		39.91	
Landing Gear & Flight Controls	17.16		17.16	
Other Systems	25.81		25.81	
Total		107.40		82.88
Age Development & Procurement for R&D		3.50		3.50
Test Aircraft & Refurbishment		34.10		34.10
Flight Test Operations		25.40		25.40
Technical Data		5.15		5.15
Total (excluding Tooling)		279.70		279.70

TABLE 5.3-4

TOTAL COSTS

Cost Element	Cost, Million $			
	Initial Studies		Present Studies	
Shop, QC, & Material Cost for 250 AC		2533.00		2611.00
Sustaining Engineering		197.65		197.65
Rate Tooling				
Wing, Tail, Fuselage, Nacelles	9.67		73.16	
Landing Gear & Flight Controls	2.59		2.59	
Other Systems	3.88		3.88	
Total		16.14		79.63
Tool Maintenance				
Wing, Tail, Fuselage, Nacelles	33.35		72.09	
Landing Gear & Flight Controls	8.88		8.88	
Other Systems	13.36		13.36	
Total		55.59		94.33
Technical Data		43.70		43.70
Warranty Expense		105.00		105.00
Basic Tooling				
Wing, Tail, Fuselage, Nacelles	64.48		39.91	
Landing Gear & Flight Controls	17.14		17.14	
Other Systems	25.78		25.78	
Total		107.40		82.88
Non-Recurring R&D		279.70		279.70
Total Program		3338.18		3493.89
Average Cost	13.35		13.98	

SECTION 6

CONCLUSIONS AND RECOMMENDATIONS

6.1 CONCLUSIONS

Additional detail has been developed for design concepts, structural weights, manufacturing approach, and expected manufacturing costs for a transonic long-range transport with extensive advanced composite applications.

For this transport the design concepts will employ very large pieces of structure, the manufacturing approach will utilize automated and mechanized fabrication procedures extensively, and both design concepts and manufacturing approach will be substantially different from those incorporated in conventional metal structure.

Alternate cost estimating procedures have been utilized to estimate the expected manufacturing costs of the airframe structure. These estimates have identified further differences in sources of manufacturing costs for composite manufacturing when compared to conventional metal manufacturing. More importantly, these new estimates have reinforced the credibility of earlier cost estimates which yielded projections of significant cost benefits to users of these advanced composite airframes.

6.2 RECOMMENDATIONS

These studies have utilized design concepts and manufacturing approaches which have not yet been developed or put into practice. Developmental activities should be initiated which couple design and manufacturing interests and have low cost composite structures as their goal. Items which should be evaluated and developed include the manufacturing of very large pieces, automated and mechanized fabrication approaches, low flow and low temperature curing materials, and structural aspects of cocured sandwich panels with non-symmetrical facings.

The cost analyses had very little prototype historical cost base and no production cost base for the graphite composite materials expected to be used on the transport. All on-going and future hardware programs should be required to develop and document manhour cost experiences in detail. In addition, specific programs designed to determine the manufacturing costs associated with repetitive manufacture of large advanced composite components should be initiated. These manufacturing experiences should utilize mechanized manufacturing procedures.

These studies have verified and expanded earlier ATT systems studies findings. Those hardware development programs identified by the earlier findings and designed to provide confident experience in applications of these advanced composite materials should be inaugurated.

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